WADD TECHNICAL REPORT 60-651
PART III

ADVANCED FLIGHT VEHICLE SELF-ADAPTIVE FLIGHT CONTROL SYSTEM

Part III Support Equipment Study

C. B. Exby
J. R. Gray
J. W. Harrison

Minneapolis-Honeywell Regulator Company
Aeronautical Division
Minneapolis, Minnesota

24 February 1961

FLIGHT CONTROL LABORATORY

Contract No. AF33(616)-6610
Project No. 8226
Task No. 10889

WRIGHT AIR DEVELOPMENT DIVISION
AIR RESEARCH AND DEVELOPMENT COMMAND
UNITED STATES AIR FORCE
WRIGHT-PATTERSON AIR FORCE BASE, OHIO

NOTICES

When Government drawings, specifications, or other data are used for any purpose other than in connection with a definitely related Government procurement operation, the United States Government thereby incurs no responsibility nor any obligation whatsoever; and the fact that the Government may have formulated, furnished, or in any way supplied the said drawings, specifications, or other data, is not to be regarded by implication or otherwise as in any manner licensing the holder or any other person or corporation, or conveying any rights or permission to manufacture, use, or sell any patented invention that may in any way be related thereto.

☆

Qualified requesters may obtain copies of this report from the Armed Services Technical Information Agency, (ASTIA), Arlington Hall Station, Arlington 12, Virginia.

☆

Copies of WADD Technical Reports and Technical Notes should not be returned to the Wright Air Development Division unless return is required by security considerations, contractual obligations, or notice on a specific document.

AF-WP-B-MAY 60 1M

FOREWORD

This report was prepared by the Aeronautical Division of the Minneapolis-Honeywell Regulator Company, Minneapolis, Minnesota under Air Force Contract AF33(616)-6610, Supplemental Agreement No. 2, under Task No. 10889 of Project No. 8226, Advanced Flight Vehicle Self-Adaptive Flight Control System. The work was administered under the direction of the Flight Control Laboratory, Wright Air Development Division. Lt. Thomas C. Hays was the task engineer for the Laboratory.

Parts I and II of Technical Report 60-651 cover the study and design phases, respectively, of the Advanced Flight Vehicle Self-Adaptive Flight Control System program. These reports are classified CONFIDENTIAL and may be obtained by qualified organizations from the Armed Services Technical Information Agency (ASTIA).

The self-adaptive flight control system is being designed to operate in advanced aerospace vehicles and will be flight tested in the X-15 in late 1961.

This report is designated by Minneapolis-Honeywell as MH Aero Report 2373-TR3.

ABSTRACT

This report presents the results of the design study to develop the ground support equipment for the MH-96 Adaptive Flight Control System to be tested in the X-15 aircraft. Design solutions are presented which reflect the problem areas that became apparent as the designs of the ground support equipment progressed. Since the final flight control system design changes have not yet been made, the GSE solutions have also not been finalized. However, this report presents a summary of the design practices being followed in the development of the ground support equipment for the MH-96 Adaptive Flight Control System.

PUBLICATION REVIEW

This report has been reviewed and is approved.

FOR THE COMMANDER:

ALFIERO J. LONGIARU
Acting Chief, Control Synthesis Branch
Flight Control Laboratory

CONTENTS

		Page
SECTION I	INTRODUCTION	1
SECTION II	X-15 GSE CONSIDERATION	2
	Flight Test Program	2
	Platform and Q-Ball Inputs	2
	Ballistic Control Rockets	3
	Stability Augmentation System Rate Gyros and Servos	4
	Mockup Bench and Flight Line Analyzer Combination	5
	NASA Instrumentation Pickoffs	5
	Umbilical Connections	5
	Component Testing	6
SECTION II	MH-96 FCS SUPPORT EQUIPMENT CONSIDERATIONS	8
	Flight Line Analyzer	10
	Airborne Checkout	25
	Accelerometer Testing Boom	46
	Redundancy and Failsafety	46
	Self-Test Procedures	49
	Environmental and Life Testing	51
SECTION IV	EQUIPMENT DESCRIPTION	56
	DUG1120A1 Flight Line Analyzer	56
	DUG1331 Airborne Analyzer	59
	DUG1352 Bench Mockup	62
	DUG1345 Accelerometer Testing Boom	64

ILLUSTRATIONS

Figure		Page
1	Computer Control Panel	13
2	Pitch Axis Simulation	19
3	Lateral Axis Simulation	21
4	Conventional Method of Closed-Loop Testing	29
5	A New Method of Closed-Loop Testing	31
6	DUG1331 Airborne Analyzer Outer Loop Simulation Redundant Channels	37
7	DUG1331 Airborne Analyzer - Block Diagram Pitch Axis Checkout	40
8	Simplified MH-96 FCS Yaw Axis	48
9	Accelerometer Self-Test	50
10	Rate Gyro Self-Test	52
11	Servo and Aircraft Simulation	55
12	DUG1120 Flight Line Analyzer	57
13	Layout of Flight Line Analyzer Modules	58
14	DUG1331 Operator's Control Unit	60
15	DUG1331 Switching Unit	61

TABLES

Table		Page
1	Flight Line Analyzer Closed-Loop Simulation Limits	14
2	Pitch Axis Equations	18
3	Lateral Axis Equations	20
4	Test Procedure	41

SECTION I
INTRODUCTION

This report covers the design problems and solutions associated with developing the ground support equipment (GSE) for an advanced flight control system of a self-adaptive nature. This system, hereafter referred to as the MH-96 FCS, is under development by the Aeronautical Division of the Minneapolis-Honeywell Regulator Company under sponsorship of the Flight Control Laboratory, Wright Air Development Division, on Contract AF33(616)-6610. A description of this system is included in the final report of the WADD-sponsored study to design an advanced self-adaptive flight control system suitable for high-altitude space craft (WADD Technical Report 60-651 Part II).

Specific problems arose in the GSE design program that were related to the nature of the X-15, in which the self-adaptive flight control system will be tested, and to the nature of the MH-96 FCS. Sections II and III cover these areas, with Section III further subdivided into MH-96 FCS support equipment considerations on the ground, in the air while the X-15 is still attached to the B-52 launch aircraft, and during environmental and life tests. Section IV contains a description of the current support equipment designs.

Manuscript released by the authors 24 February 1961 for publication as a WADD Technical Report.

SECTION II
X-15 GSE CONSIDERATION

FLIGHT TEST PROGRAM

The approach taken in designing support equipment for the X-15 program was to make it as versatile as possible and to present maximum information to the GSE operator, within practical design considerations. This approach results from the nature of the X-15 program and the advanced flight control system to be installed in this aircraft.

The nature of the flight test program planned for the MH-96 FCS and the few flights allocated to proving the performance of this system require a maximum effort to ascertain correct system status parameters during ground checks. This concept is further emphasized by the intended use of the flight control system during flights to extreme altitudes and re-entry angles where loss of the flight control system could possibly be catastrophic.

The MH-96 FCS support equipment was designed for the highly skilled Air Force and NASA maintenance personnel on the X-15 program. Since maintenance will never be delegated to semi-skilled personnel, automatic ground checkout is not justified on this basis. However, the limited time available for last-minute airborne checks of the MH-96 FCS and the comprehensive life tests dictate the automation of these tests. The reasons for the choice of automatic airborne and life test equipment are developed in Section III.

PLATFORM AND Q-BALL INPUTS

Honeywell was requested to formulate an over-all ground support equipment maintenance philosophy for the MH-96 FCS. This was to include the integration of existing ground support equipment for such systems as the X-15 inertial platform and dynamic pressure sensor or "Q-Ball" into the specific support equipment requirements for the MH-96 FCS. Analysis of the support equipment for these systems showed that complete systems tests (that is, those tests where test signals originate in the prime sensors) were impractical because of the difficulties in stimulating the existing inertial platform or Q-Ball system from an FCS test set. For this reason, the closed-loop testing of the adaptive flight control system was restricted to exclude these devices. These outer-loop holding modes are simulated by the GSE analog computer rather than by being stimulated by a test set. To check these system inputs before each flight, the X-15 preflight procedure must have a measurement test whereby specific FCS input voltages are recorded for specific attitudes, headings, and angles of attack from these sensors. The only parameter not measured is the dynamic response of these sensors. However, because of the thorough testing of these systems by test operators using specialized checkout equipment, this parameter can be reasonably assumed to have been adequately checked.

BALLISTIC CONTROL ROCKETS

The extreme altitudes for some X-15 mission profiles require the use of ballistic control rockets by the MH-96 FCS. These rockets are solenoid-valve-controlled and use a monopropellant of hydrogen peroxide. The catalyst beds in these rockets are the critical factor in determining the useful life of the units, and their status prior to flight is extremely critical. The degree of clogging and scaling of these beds is of prime importance in determining the acceptable performance characteristics when they are used with the automatic flight control system. The best indication of the condition of these beds, short of dismantling the valves, is to measure the chamber pressure during the ground checkout of the MH-96 FCS. The Air Force is installing pressure transducers in the rocket motors of the X-15 No. 3, in which the MH-96 FCS will be installed. These transducers will be used on the ground during checkout procedures and during flight for instrumentation purposes. The GSE includes strain gage preamplifiers to record the chamber pressure (related to thrust) on an oscillograph.

An additional justification for the inclusion of a means of measuring ballistic control rocket parameters is that the expected life of these rocket motors, because of catalyst bed considerations, is estimated to be between 15 and 30 minutes. There is no record of the actual duty cycle or average life of the rockets because flights requiring their use have been infrequent in the X-15 flight test programs up to this time.

STABILITY AUGMENTATION SYSTEM RATE GYROS AND SERVOS

It was decided by WADD and Honeywell that all suitable components of the X-15 aircraft should be utilized. Therefore, the rate gyros and servos of the stability augmentation system (SAS) were retained, and the support equipment for these components was considered adequate. This equipment consists of a rate table console with associated gyro mounting fixtures and a hydraulic flow bench with associated servo test fixtures. No new equipment was required for dynamically testing rate gyros and servos, and a subsequent cost savings was accomplished for the support of the MH-96 FCS.

The rate gyro packages installed in the X-15 for the SAS system provided an excellent means for obtaining self-test modes in these units. The SAS system provided gyros with two signal pickoff windings per gyro. Both pickoffs were used in the SAS system, one for control and one for failsafe monitoring. Because of the redundancy provisions in the MH-96 FCS, where two separate gyro packages are used in each axis, it was possible to utilize one of the windings on each of the six rate gyros as a torque input. Thus the gyro could be tested while installed in the aircraft -- and even while airborne and attached to the B-52 aircraft -- by applying a calibrated current input to each gyro torquing coil and evaluating the signal outputs corresponding to a specific rate. This test could be performed without interrupting any signal or power circuits to the gyro. No modifications were made to the basic SAS gyro as a result of this use.

MOCKUP BENCH AND FLIGHT LINE ANALYZER COMBINATION

Considering the intended use of the support equipment at Edwards AFB, California, it was decided to incorporate the proposed field-type mockup bench and the proposed flight line analyzer into one unit in order to reduce costs. To achieve the required mobility of the MH-96 FCS support equipment, a single four-wheeled trailer was designed both for flight line use and for bench mounting and trouble shooting.

NASA INSTRUMENTATION PICKOFFS

To reduce cost and airborne weight, it was decided to use the existing NASA instrumentation potentiometers for control surface position rather than to install individual GSE surface position transducers. Information of the actual surface position, rather than servo position which is readily available from existing servo pickoffs, is required to close the loop accurately during the ground testing of the MH-96 FCS and to include the inherent actuator hysteresis and dead spots within the simulation. NASA potentiometers Nos. 28, 32, and 34 (NAA Nos. 1, 3, and 4) were used to measure the upper vertical stabilizer, the right horizontal stabilizer, and the left horizontal stabilizer positions. These potentiometer leads will be brought to a disconnect assembly in the elevator area so that they may be used for ground checkout of the MH-96 FCS and for airborne recordings.

UMBILICAL CONNECTIONS

The umbilical connections from the B-52 to the X-15 aircraft do not contain any spare pins for use in performing prelaunch confidence tests on the MH-96

FCS prior to launching. Therefore, it was necessary to install a new connector both in the B-52's and in the X-15 selected to be equipped with the MH-96 FCS. No other test connectors except those mounted on the MH-96 FCS adaptive calibrator are utilized. The 110 umbilical connector wires lead from the B-52 airborne analyzer through the pylon and into the X-15 adaptive controller via two test connectors only. Thus when the two aircraft part at launch, no system loads are broken (all umbilical connections are through test points only), and the possibility of disturbing the system at launch is decreased. Where possible, this concept of parallel testing will be accomplished in all levels of maintenance of the MH-96 FCS, both in the air and during ground checkout.

COMPONENT TESTING

Owing to the limited number of adaptive systems being fabricated for the X-15, it was decided at the beginning of the program that no individual printed circuit card test sets would be constructed for use at the flight test facility at Edwards AFB. Spare printed circuit cards will be provided so that the test program will not be held up because of a failure of these components. Spare printed circuit cards will be purchased in limited quantity, and all major repairs or modifications to these cards will be accomplished in Minneapolis. A two-week turn-around time is assumed.

At Edwards AFB, minor repairs and checks of individual card performance will be made with in-system tests or with the individual card removed from the adaptive controller and powered by the flight line analyzer. Standard power supplies for the respective printed circuit cards will be provided within the flight line analyzer, along with suitable physical and electrical mounting facilities. Loads and interconnections peculiar to individual card types will not be provided as an integral part of the analyzer but will be easily mocked up with resistors, capacitors, and other parts normally present in an aircraft electronic maintenance shop. Most of the standard instrumentation necessary to ascertain the response characteristics will be readily available in the analyzer which contains an oscilloscope, a d-c vacuum-tube voltmeter, an a-c vacuum-tube voltmeter, a low-frequency function generator, and an oscillograph. This component maintenance concept was mechanized in the flight line analyzer, resulting in a reduction in the number and types of test sets required for the support of the MH-96 FCS.

SECTION III
MH-96 FCS SUPPORT EQUIPMENT CONSIDERATIONS

The MH-96 FCS is an advanced self-adaptive flight control system of a type which can be applied to a variety of vehicles for control throughout mission profiles involving wide ranges of altitude and speed. The support equipment for checking such systems is unique in that closed-loop testing is a necessity inherent in the reliability requirements for hypersonic vehicles. The dynamic characteristics of the vehicle and the FCS together establish the flight characteristics during the critical re-entry phase, and closed-loop testing is the only way to properly evaluate the FCS and vehicle performance.

The requirement for closed-loop testing is in addition to open-loop testing such as is normally required for checkout of conventional linear flight control systems.

Since very few flight tests of the MH-96 FCS in the X-15 aircraft will be made, only a short time is available for proving system performance. Consequently, since drops from the B-52 mother ship must be made with complete confidence in system functional readiness, pre-drop tests must be made with unusual thoroughness. In this case, closed-loop testing is employed both on the ground and in the air immediately preceding each drop. The analog simulation of the response of the X-15 used for closing the outer loop about the FCS during ground checkout is fairly elaborate. The electronic simulation used during the airborne checks is, of practical necessity, much simpler.

The support equipment requirements for adaptive systems are dependent on the particular system and its operational concept. The complexity of the simulations necessary for closed-loop testing will vary from system to system.

In some instances it may be known that system performance is not strongly dependent on flight conditions; a single vehicle response simulation involving fixed parameters may then suffice to check the system adequately. An example of such a case might be the control system for a missile booster, as contrasted with a complete orbital re-entry type of manned vehicle where much more severe ranges of altitude and dynamic pressure would be encountered.

The operational concept of the vehicle is also important in determining the scope of the testing program and the complexity of the support equipment. For a program in which the study of system performance at various flight conditions is a prime objective, the support equipment may be required to predict performance or derive data for correlation with flight test results as well as to verify system readiness. In this case, full six-degree-of-freedom analog simulations of the vehicle response are conceivable.

Where the purpose of the support equipment is merely to perform routine periodic functional checks on a system, very simple response simulations may be adequate. In many cases these simulations need not be representative of any particular flight condition and need not closely represent the actual aircraft providing the response of the adaptive system under these conditions is adequately defined for a known good system. It must also be known, of course, that the arbitrary simulation employed will react in an easily recognized fashion to all possible failures of the FCS which would preclude its use with the actual aircraft. It is apparent that this method in support equipment design presupposes a history of FCS performance both of the actual aircraft (or of some true counterpart) and with the arbitrary simulation.

Additional considerations contributing to a definition of the specific support equipment design requirements for adaptive systems are (1) the degree of reliability required, (2) practical economy involving the number of systems to be checked and the frequency of checks, (3) the time limits set for checks, and (4) the technical level of personnel using the equipment. Support equipment for adaptive systems could be digitally automated in a manner similar to that now being used in several instances for checking conventional flight control systems from punched tape programs. A further possible refinement might be a requirement for continuous monitoring of performance over specific mission profiles. In such a case, it could be necessary to program the simulation of the vehicle with time for the flight conditions encountered as the vehicle progresses. This would involve the continuous variation of the stability derivative coefficients appearing in the equations of motion for the vehicle. The analog mechanization of these changes would be extremely complicated because the coefficients vary along complex curves, becoming alternately positive and negative in sign. Calibrated cam-actuated potentiometers would probably be necessary, with special consideration being required to implement the sign changes.

It is apparent from the above discussion that the essential distinction between support equipment for adaptive systems and for conventional linear control systems is the requirement for closed-loop testing. The manner in which this is accomplished and tests to be performed in this mode depend on the specific application. Therefore, further generalizations are of little value. Consequently, the following discussion deals specifically with the development of support equipment for the MH-96 FCS.

FLIGHT LINE ANALYZER

FUNCTIONS

The flight line analyzer provides the following functions: (1) support instrumentation for the X-15 flight test program, (2) verification of flightworthiness of the MH-96 FCS, and (3) fault isolation of components in the FCS. These three objectives dictate the physical configuration, content, and general design philosophy of the device.

The flight line analyzer is designed especially for the X-15 research program. It is a one-of-a-kind device and should properly be designated as an engineering developmental model. It is designed for use by personnel of relatively high technical skill, yet the convenience of the operator has been a prime consideration. While reliability has been emphasized, it was necessary in some instances to incorporate commercial quality components. Critical components are supplied with spares. Every effort has been made in the layout of the electrical circuits to prevent failures due to inadvertent misuse on the part of the operator. The number of adjustments and external electrical interconnections which the operator must make in exercising the various checks has been kept to a minimum.

Support Instrumentation

An important aspect of the flight line analyzer is its ability to be used as a research tool in support of the X-15 flight test program. In effect it is possible, by means of the analog computer and the eight-channel recording facilities incorporated into the analyzer, to operate the FCS under many simulated conditions of X-15 flight. Undoubtedly in-flight tests will reveal many areas which require closer study, not only to predict probable performance but to assist in analysis of observed behavior. The Electronics Associates Inc. PACE TRIO Analog Computer provides thirty operational amplifiers, twenty-eight coefficient potentiometers, and ten integrator modules. These components are sufficient to permit a very close simulation of the response of the X-15 to the MH-96 FCS under various flight conditions.

In its normal usage, the computer components are preprogramed on patch cards contained in drawers of the analyzer that are separate from the computer. Input and output connections to all components in the computer are brought out in electrical cables at the rear of the computer for interconnection with the associated analyzer circuitry. These same points are duplicated in a patch board array on the front of the computer and are available to the operator at the flight line analyzer console. A hinged door normally covers this external patch board and prevents the operator from attempting to program the computer while it is preprogramed within the analyzer.

The computer assembly is mounted on slides in the analyzer. By withdrawing the computer, the connections at the rear can be broken to isolate the computer from the analyzer circuits and permit it to be used as an independent tool for implementing any simulation. In this case, the door on the front patch board of the computer is opened to permit programing. Inputs to the computer from the FCS and outputs from the computer to the recording equipment are accomplished by means of external leads connected between the appropriate jacks on the computer patch board and on the main operator's control panel on the console of the analyzer. Both a-c and d-c vacuum-tube voltmeters are available on the main panel for monitoring voltages.

In Figure 1, the control switching for the computer in its preprogramed mode is shown. In the preprogramed mode, the option of varying the coefficient potentiometer settings of the computer to simulate various flight conditions is allowed. Not all sign configurations of the stability derivative coefficients appearing in the equations of motion can be handled, but a great many flight conditions can be simulated by means of the variable coefficient potentiometers and the switching provided.

The panel shown in Figure 1 is incorporated into the computer mounting assembly and appears on the console of the flight line analyzer. The coefficient selector switch shown in Figure 1 permits the simulations of nine different variable coefficient sign configurations. The switch positions, together with a table of coefficient values, will accomplish the simulation of 34 of the 37 flight conditions studied in the very elaborate simulation used during the design of the MH-96 FCS. These 34 flight conditions are listed in Table 1. These sign configurations will undoubtedly be applicable to many other flight conditions. Where they are not applicable, the computer can be programed externally as described above to accomplish the necessary sign changes. The use of the analyzer and computer in this manner requires, of course, a highly skilled operator.

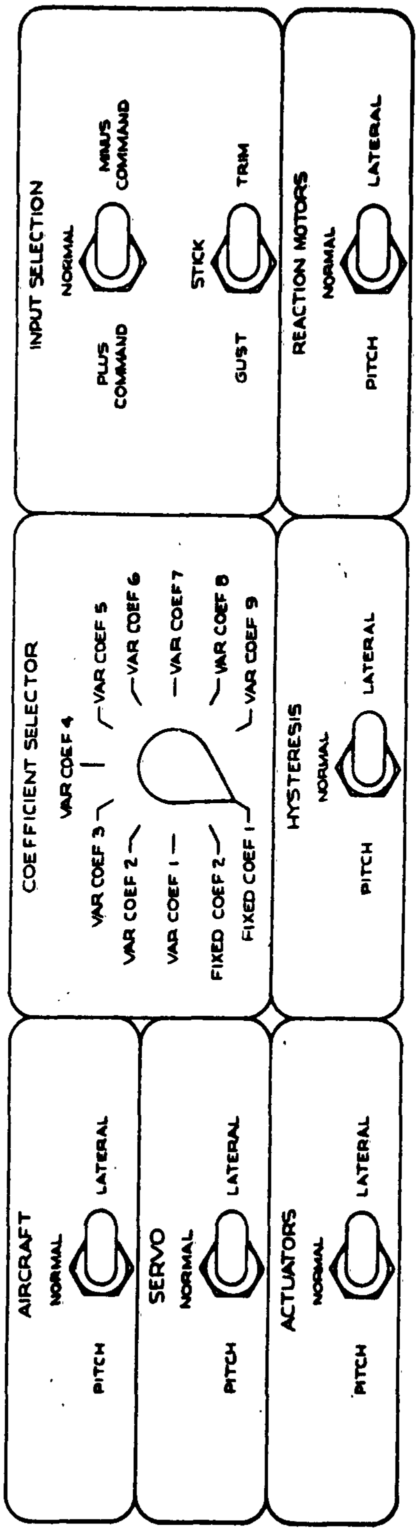


Figure 1. Computer Control Panel

Table 1. Flight Line Analyzer Closed-Loop Simulation Limits

Flight Condition	Mach No.	Altitude (feet)	Angle of Attack (degrees)
1	0.6	35,000	18.6
2	0.8	40,000	11.6
3	0.8	40,000	20.0
4	1.0	40,000	6.4
5	1.0	40,000	20.0
6	1.5	50,000	0.0
9	2.0	70,000	0.0
10	2.0	70,000	8.2
11	3.0	90,000	0.0
12	3.0	90,000	10.8
13	4.0	100,000	0.0
14	4.0	100,000	10.3
15	5.0	130,000	0.0
16	6.0	140,000	0.0
17	6.0	120,000	0.0
18	6.0	120,000	11.5
19	6.0	100,000	5.3
20*	6.0	100,000	30.0
22	5.0	100,000	6.4
23	4.0	100,000	8.2
24	3.0	100,000	11.4
25	2.0	80,000	7.4
26	1.5	70,000	6.6
27	1.2	40,000	2.0
28	1.2	10,000	0.5
29	1.0	10,000	1.0
30	0.6	10,000	2.6
31	0.6	5,000	2.0
32	0.2	0	17.0
33	6.0	160,000	0.0
34	6.0	180,000	0.0
35*	6.0	200,000	0.0
36	6.0	220,000	0.0
37	6.0	280,000	0.0

*Preflight conditions

Verification of Flightworthiness of the FCS

Ground checkout of the MH-96 FCS is envisioned in two stages: comprehensive periodic checks, and preflight checks prior to each mission. The preflight checks will augment, and in many cases duplicate, the airborne checks made just prior to drop. The preflight checks will involve only the necessary minimum number of open-loop and closed-loop tests.

Two flight conditions, representing both the aerodynamic and ballistic regimes of flight, were selected for simulation during the closed-loop tests. The selection was based on their severity in terms of demand on the performance of the FCS. These two conditions are denoted by asterisks in Table 1 and are selected by turning the coefficient selector switch illustrated in Figure 1 to the first two fixed coefficient positions.

In the closed-loop preflight checks, it is probable that only the aircraft response and the dynamics of the reaction motors will be simulated. Hydraulic power will be required so that all servos and actuators may be checked. The X-15 will not be attached to the B-52 mother ship during these preflight ground checks.

The preflight checks will normally consist of those checks for which parallel testing is possible; i.e., no system connections will be broken for insertion of the ground support equipment. Normally, in the preflight checkout, the connection of the flight line analyzer to the FCS will be accomplished through four GSE connectors provided on the BG-183 calibrator.

The comprehensive ground checks will involve open-loop and closed-loop tests of a scope sufficient to verify the performance of each component in the FCS. Both parallel and intercept testing will be employed. Closed-loop tests will be made at many or all of the permissible simulated flight conditions. Checks will be made with and without hydraulic power. In making checks without hydraulic power, the servos and actuators will be simulated by the computer in the analyzer.

Fault Isolation and Maintenance

In the event the FCS fails a particular check during either the preflight or comprehensive tests, it will be possible for the operator to execute a trouble-shooting procedure for locating the least replaceable component at fault. This may be accomplished using only the switches and meters provided on the operator's panel of the flight line analyzer. When the fault is traced to a certain printed circuit card of the adaptive calibrator, a card tester supplied as an accessory to the flight line analyzer may be used to further trouble shoot the card and check out its performance after repair.

The card tester will normally be stored in the analyzer and will derive its input power from a suitable connector provided on the operator's panel. A Tektronix oscilloscope, suitable for transistor circuit analysis, is provided in the console of the analyzer to assist card maintenance and repair.

CLOSED-LOOP TESTING

Closed-loop testing in the case of linear flight control systems, while often used during the development stages of a system, is seldom used as a method of performance inspection in the field. Open-loop testing normally is sufficient to verify performance. With adaptive systems, however, some form of closed-loop testing is mandatory; and it represents a powerful tool. The simplicity which it introduces into evaluating procedure tends to offset the greater cost involved in the mechanization of the simulations required to close the outer loop around the FCS.

By "flying" the system on the ground, it may be quickly ascertained whether the system is or is not controlling the aircraft properly in its various modes of flight. An immediate check on the system as a whole is thereby permitted; and if no malfunction is detected, many more detailed checks can perhaps be foregone. It is recognized, however, that an adaptive system may tend to be more tolerant of small departures from the specified performance of certain of its components, tending to mask their malfunctions. In the case of the MH-96 with its redundant features, this masking tendency is still more prevalent. Thus it is always necessary to perform special tests designed to point out such malfunctions. These tests may involve both open- and closed-loop testing.

The degree of simplification of the required system evaluation which closed-loop testing imparts may also depend on the extent to which the closed-loop simulation represents the actual aircraft. Closed-loop testing of the MH-96 by the DUG1120 is conducted in two modes - pitch and lateral. In the pitch control mode, the response of the X-15 is simulated about the pitch axis only. In the lateral control mode, the aircraft response is simulated about both the roll and yaw axes simultaneously. The computer components used to mechanize these simulations are shared between these two modes, and the particular mode to be checked is selected by means of the switches on the computer control panel shown in Figure 1.

The desirable feature of interlocking the switches on the computer control panel to prevent partial selection of both modes at the same time was considered but not incorporated because of the complication it represented. It will be necessary for the operator to inspect his setup carefully for such discrepancies. No damage would result from such an occurrence, but the recorded results would be meaningless. The use of two modes of closed-loop testing is permissible in the case of the X-15 since it has been determined in previous simulations at Honeywell that these control modes are reasonably independent. The ability to share computer components between these modes represents a substantial cost saving in the design of support equipment.

Honeywell investigations have shown that the linearized short-period equations of motion are adequate in representing the X-15 dynamics for mechanization in the flight line analyzer. These equations and their associated simulation diagrams are given in Table 2 and Figure 2, respectively, for the pitch mode and in Table 3 and Figure 3, respectively for the lateral mode. In the complete simulation, sensor dynamic have been ignored.

Table 2. Pitch Axis Equations

$$\dot{\alpha} = Z_w \alpha + \dot{\theta} + \frac{Z_{\delta_a}}{U_1} \delta_a + \dot{\alpha}_{RC}$$

$$\ddot{\theta} = M_\alpha \alpha + M_q \dot{\theta} + M_{\delta_a} \delta_a + \ddot{\theta}_{RC}$$

$$N_x = \frac{U_1}{57.3g} (\dot{\theta} - \alpha) + \frac{x}{57.3g} \ddot{\theta}$$

α = angle of attack

δ_a = stabilator deflection

M_α = change in pitch moment per unit pitch moment of inertia per unit angle of attack

M_q = change in pitch moment per unit pitch moment of inertia per unit pitch rate

M_{δ_a} = change in pitch moment per unit pitch moment of inertia per unit stabilator deflection

Z_w = change in force along Z axis per unit aircraft mass per unit angle of attack rate

Z_{δ_a} = change in force along Z axis per unit aircraft mass per unit stabilator deflection

U_1 = steady-state forward velocity

N_x = normal acceleration X feet forward of center of gravity

θ = pitch attitude

U_1 = aircraft forward velocity

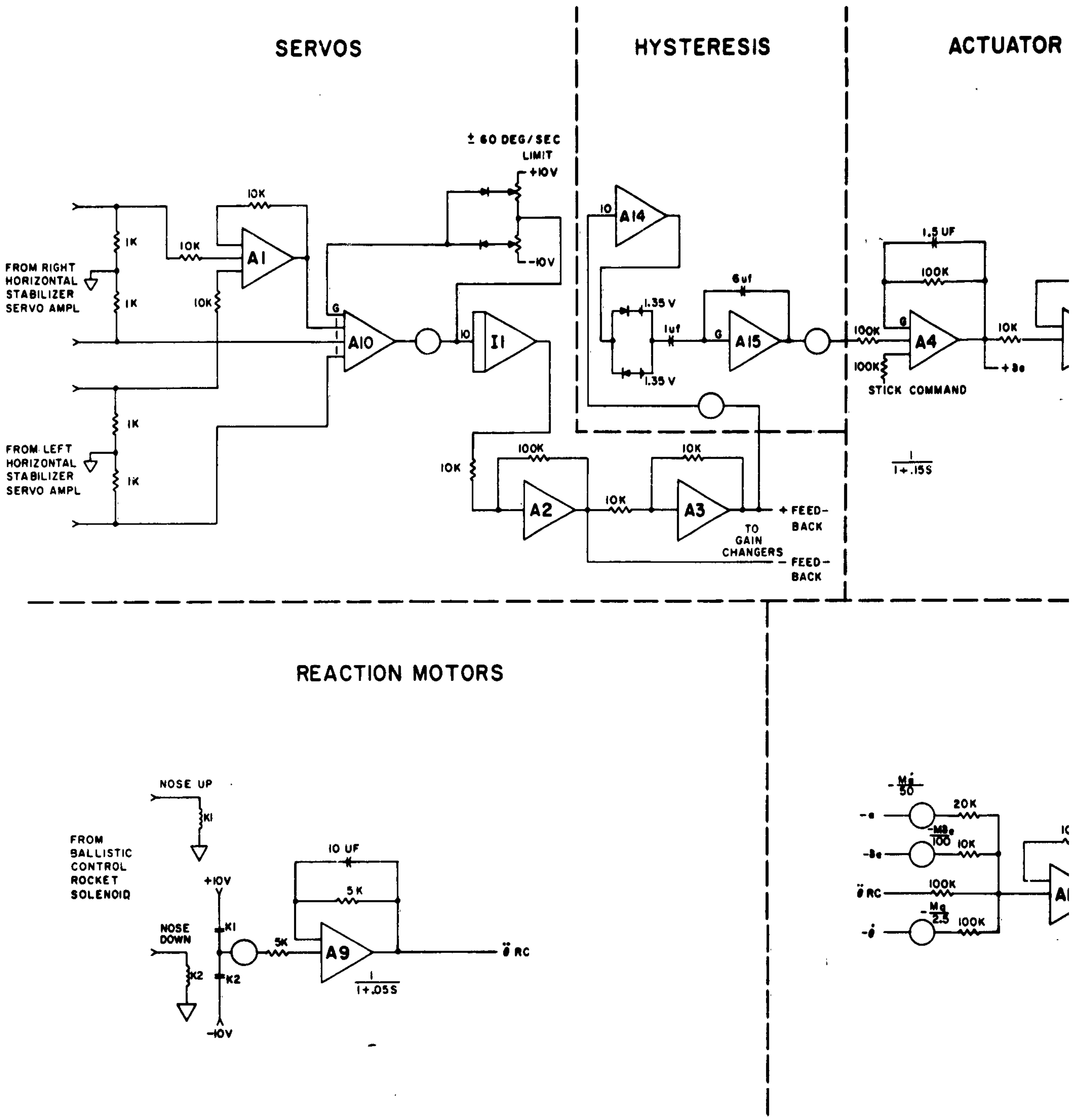


Figure 2. Pitch Axis S

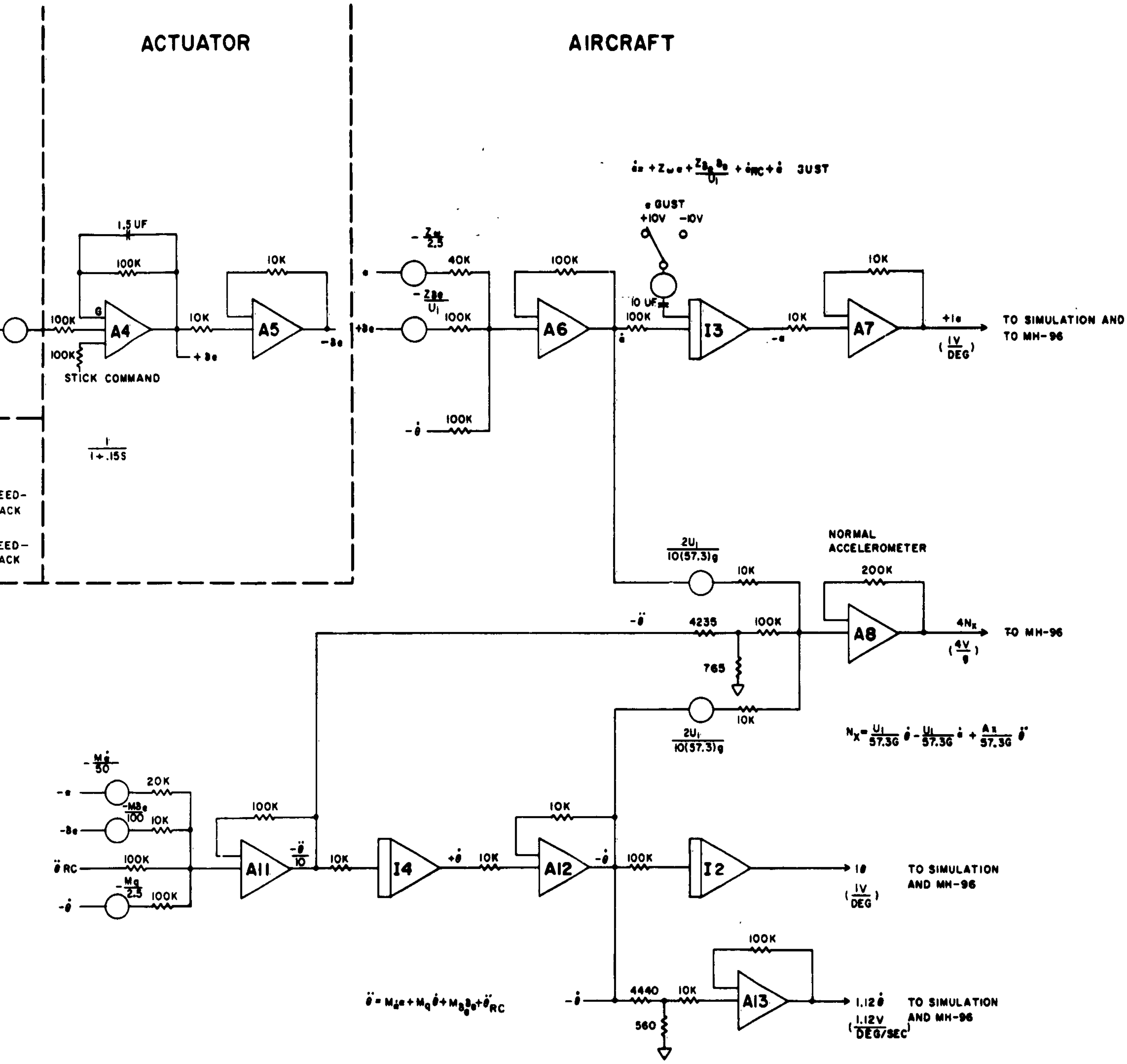


Figure 2. Pitch Axis Simulation

Table 3. Lateral Axis Equations

$$\dot{p} = L_p' p + L_\beta' \beta + L_r' r + L'_{\delta_r} \delta_r + L'_{\delta_a} \delta_a + \dot{p}_{rc}$$

$$\dot{r} = N_p' p + N_\beta' \beta + N_r' r + N'_{\delta_r} \delta_r + N'_{\delta_a} \delta_a + \dot{r}_{rc}$$

$$\dot{\beta} = Y_v \beta + \frac{g}{U_1} \phi - r + \frac{Y_{\delta_r}}{U_1} \delta_r + \frac{Y_{\delta_a}}{U_1} \delta_a + \dot{\beta}_{gust}$$

$$L_s' = \frac{L_s + \frac{I_{xz}}{I_{xx}} N_s}{1 - \frac{I_{xz}}{I_{xx}} \frac{I_{xz}}{I_{zz}}}$$

$$N_s' = \frac{N_s + \frac{I_{xz}}{I_{zz}} L_s}{1 - \frac{I_{xz}}{I_{xx}} \frac{I_{xz}}{I_{zz}}}$$

p	= roll angular rate
β	= sideslip angle
r	= yaw angular rate
δ_r	= rudder angular deflection
δ_a	= differential stabilator deflection
L_p	= change in rolling moment per unit roll moment of inertia per unit roll rate
L_β	= change in rolling moment per unit roll moment of inertia per unit sideslip angle
L_r	= change in rolling moment per unit roll moment of inertia per unit yaw rate
$L\delta_r$	= change in rolling moment per unit roll moment of inertia per unit rudder deflection
$L\delta_a$	= change in rolling moment per unit roll moment of inertia per unit stabilator deflection
N_r	= change in yawing moment per unit yaw moment of inertia per unit yaw rate
N_s	= change in yawing moment per unit yaw moment of inertia per unit sideslip angle
N_p	= change in yawing moment per unit yaw moment of inertia per unit roll rate
$N\delta_r$	= change in yawing moment per unit yaw moment of inertia per unit rudder deflection
$N\delta_a$	= change in yawing moment per unit yaw moment of inertia per unit stabilator deflection
ϕ	= roll attitude moment per unit yaw moment of inertia
Y_v	= change in force along Y axis per unit mass of aircraft per unit sideslip velocity
$Y\delta_r$	= change in force along Y axis per unit mass of aircraft per unit rudder deflection
$Y\delta_a$	= change in force along Y axis per unit mass of aircraft per unit stabilator deflection
S	= denotes an arbitrary subscript
I_{xx}	= moment of inertia about X axis
I_{zz}	= moment of inertia about Z axis
I_{xz}	= product of inertia about X and Z axis

$$N_y = Y_v \beta + \frac{Y_{\delta_r}}{U_1} \delta_r + \frac{Y_{\delta_a}}{U_1} \delta_a + K \frac{L_x}{U_1} \dot{r}_p$$

N_y	= lateral acceleration L_x feet forward of center of gravity
K	= a constant

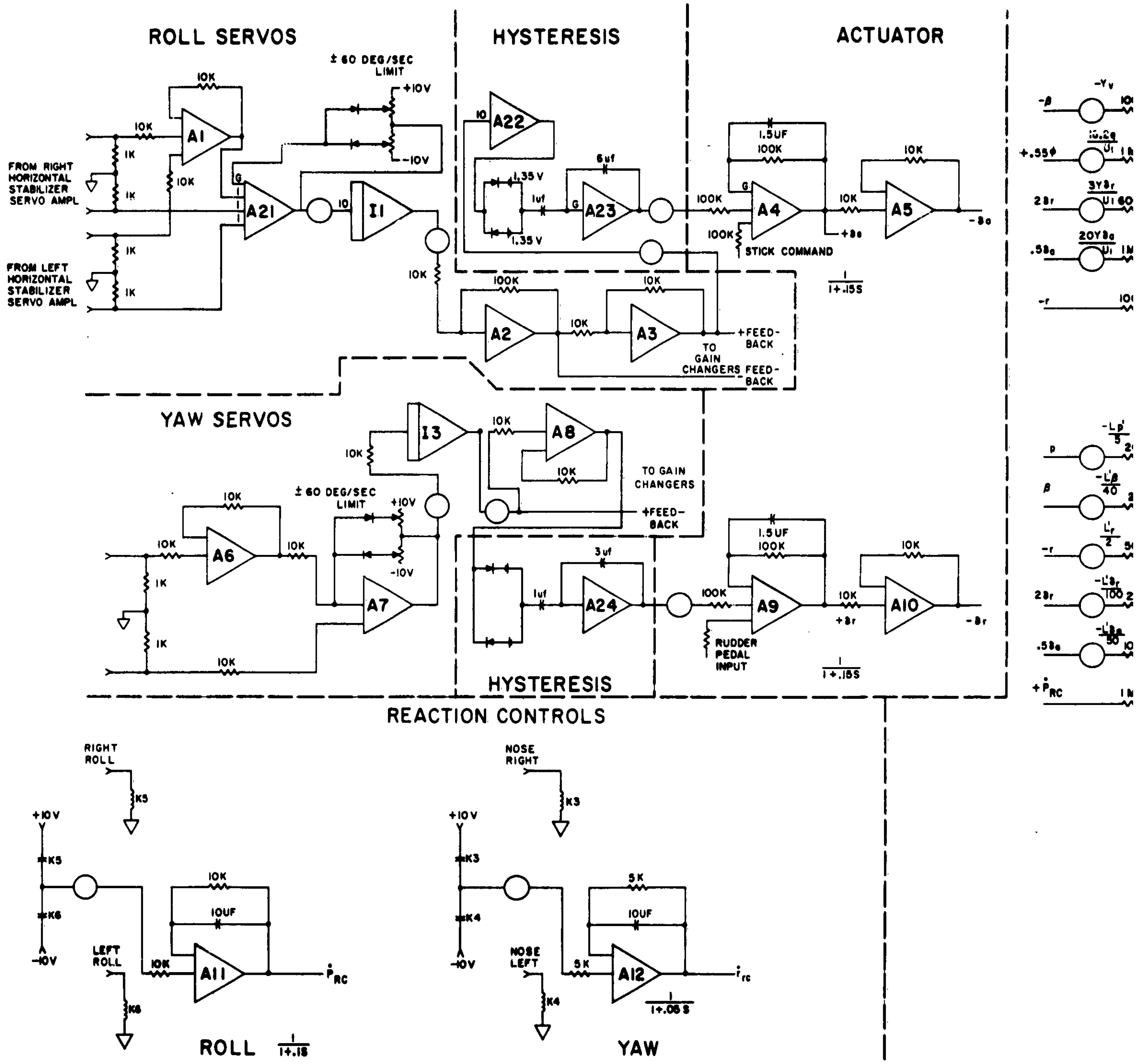


Figure 3. Lateral Axis Simulat

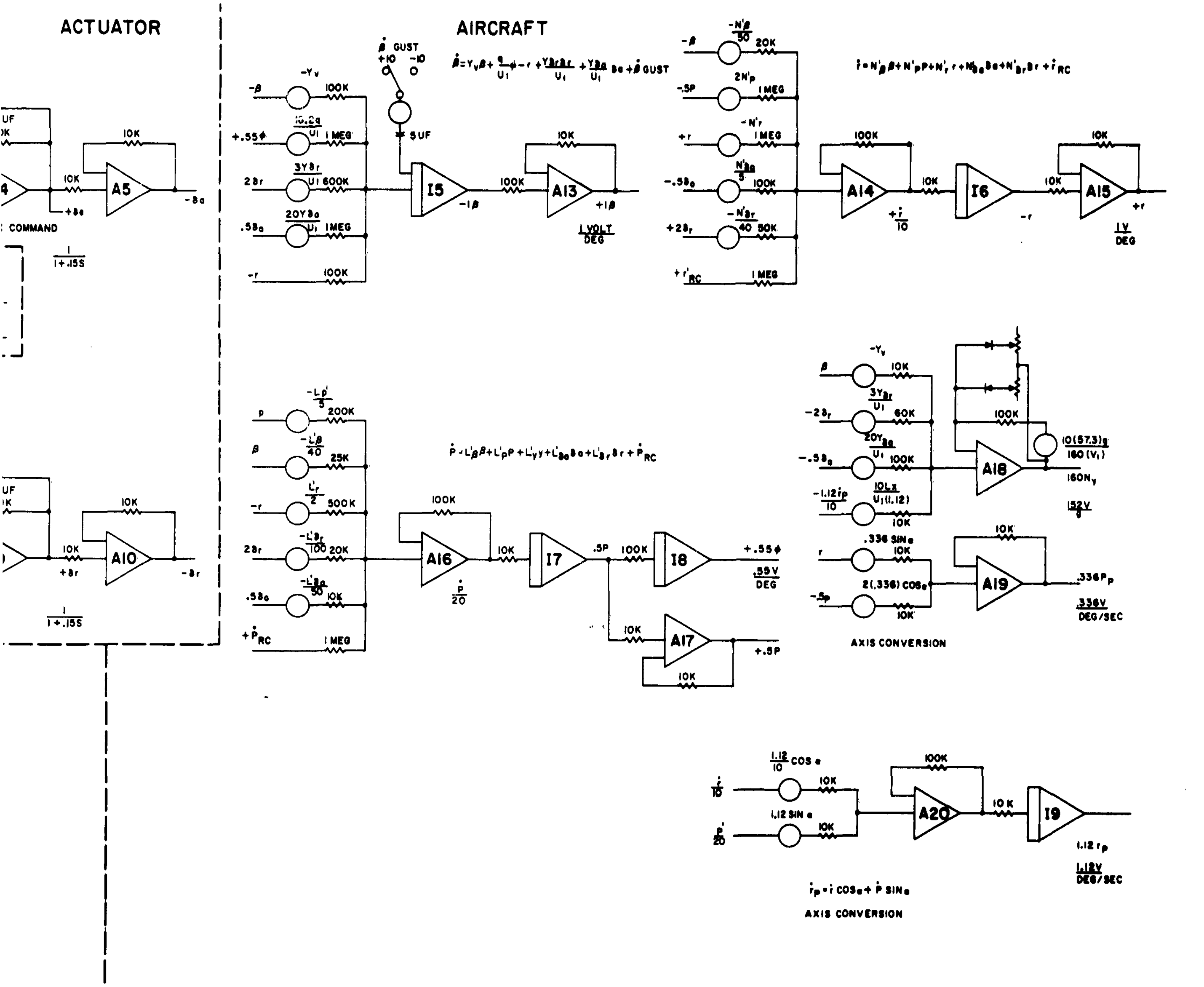


Figure 3. Lateral Axis Simulation

This has been determined to be a reasonable approximation and consistent with the use of the linearized equations. The proper static gains and scale factors for these items have been incorporated.

In simulating the response of the aircraft under various flight conditions, all of the variable coefficients which appear in the equations and which are functions of the flight conditions are changed using the potentiometers of the computer. These potentiometers must be adjusted by the operator from flight condition to flight condition except in the case of the preflight checks. Here the coefficient potentiometers are removed from the preprogramed simulation circuitry by the computer panel switching and are automatically replaced by fixed resistance voltage dividers.

All other elements of the simulation are fixed on the preprograming patch cards contained within the chassis located at the rear of the flight line analyzer. To change the fixed simulation, it is necessary to remove these patchcards. Components are connected in relation to each other by means of solder connections. This method of construction has been used to prevent the simulation circuits from being easily changed by inexperienced personnel and yet to permit their alteration if required. The use of preprogramed simulations in this manner not only relieves the operator of tediously setting up the computer, but also ensures uniformity and repeatability of the closed-loop tests.

Simulations of the pitch and lateral reaction controls, stabilator and rudder servos, and stabilator and rudder actuators are also provided. The option of including the effect of actuator linkage hysteresis is also provided. These simulations may be selected in any combination by means of the switching provided on the computer control panel. The option of selecting these additional simulations permits the flight line analyzer to be used as a bench tester for checking the calibrator when it is removed from the aircraft as well as when it is installed in the aircraft with and without hydraulic power. It is thus possible to trouble shoot the system by a process of progressive elimination in which the performance of these major components may be evaluated by appropriate comparisons.

During ground checks, the aircraft reaction motors will be de-activated. Since, in any case, they could serve no purpose in closed-loop testing, it is necessary to simulate their effect. The option of switching these simulations out of the system is provided so that comparisons can be made to ascertain the effect of the reaction controls at certain flight conditions.

In general, the closed-loop testing will involve measurements of the adaptive controller gain changes, amplitude variations of the characteristic oscillation, and response of the system to various step commands and gust inputs. These inputs will be initiated both from the main operator's control panel or from the computer control panel as shown in Figure 1. Typical output parameters which will be monitored by the recorder are angle of attack (α), pitch rate ($\dot{\theta}$), pitch attitude (θ), normal acceleration (N_z), roll rate (r), roll attitude (ϕ), yaw rate (p), yaw attitude (ψ), lateral acceleration (N_y), sideslip angle (β), stabilator deflection (δ_e), rudder deflection (δ_r), and adaptive controller gain.

The evaluation of recorded traces will be facilitated by means of transparent overlays on which the proper performance limits are printed. This will relieve the operator of the burden of interpretation.

The dynamic response of the MH-96 FCS gain changer is considered a significant parameter in determining adequate system performance, particularly during the re-entry maneuver. The test of this parameter will be accomplished by switching from one fixed flight condition to another during the preflight mode and observing the recorded trace of the gain changes in each axis. The transient response of the gain changer for increasing and decreasing gains may be checked from measurements along the time axis of the recorder trace. It was originally contemplated to drive the band-pass input to the variable gain amplifier or to excite the rate gyro inputs with a variable low-frequency sine wave signal and to record the resulting gain pertubations. However, this scheme proved incompatible with normal usage of the gain changer, where servo signals to the band pass are not necessarily pure sine wave signals in that noise distorts these feedback inputs to the band-pass amplifier.

In the planned use of the analog computer on the flight line to accomplish closed-loop testing, a problem arose associated with the difficulties of coupling a d-c computer to a flight control system using a-c sensors and feedback signals. While it was considered desirable to modulate the aircraft 400-cps response outputs for application to the adaptive controller for loop closure, the difficulties in accomplishing this far outweighed the benefits. These difficulties resulted from the null requirements imposed on the modulator for very low values of computer outputs. The sine wave modulator must be better than the analog computer in its response characteristic.

The quality of the sine-wave-modulated signal of the modulator would not result in gross errors in simulations performed on the adaptive system if sufficient design work were applied to the development of the modulator. However, when a buffer exists between the analog simulation and the system about which the loop is being closed, there is a chance for this loop closure to be an untrue replica of the intended flight condition programed by the analog computer. This is particularly true of modulators such as simple choppers where the output signal is an in-phase or out-of-phase square wave rather than the simulated sensor signal, which is a true sine wave. The response of the adaptive system amplifier-demodulators is not necessarily the same for these square wave inputs as for the normal sensor sine wave inputs.

The condition described above would deteriorate the accuracy of the simulation. Therefore, it was decided that the flight line analyzer should insert its loop closure signals directly from the d-c analog computer into the MH-96 FCS at a point beyond the system amplifier-demodulators. These units will be checked for dynamic and static gain during the open-loop testing mode, and this method should impose no problem in thoroughness of testing. Insignificant lags are associated with these amplifier-demodulators; therefore, their exclusion from the closed-loop simulation should have a negligible effect on simulation accuracy.

OPEN-LOOP TESTING

The use of closed-loop testing of an adaptive system does not remove the requirement for open loop testing. It is more convenient to observe certain transfer characteristics during open-loop rather than closed-loop tests. Individual components of the adaptive system can be easily checked for gain and frequency response as separate units. For this reason the flight line analyzer was designed to include open-loop testing capabilities.

If the variable-gain amplifiers are driven to a known gain or are resting at either the maximum or minimum gains, the adaptive flight controller may be treated as a linear servo system; and trouble shooting may be performed accordingly. The insertion of low-frequency sine or square waves may be accomplished, with the MH-96 FCS acting as a linear system, for both frequency response and transient response analysis. The flight line analyzer contains a rack-mounted function generator for this purpose.

AIRBORNE CHECKOUT

AIRBORNE CHECKOUT REQUIREMENTS

The function of the airborne analyzer is to perform a complete checkout of the X-15 FCS during a brief checkout period of approximately 2 minutes just before the vehicle is launched from the B-52. The most significant design requirements for the checkout are:

- Remote Location -- The checkout equipment is to be located in, and operated from, the B-52 launch aircraft. The checkout shall require little, if any, inputs by the X-15 pilot.

- Time Limit -- The time during which hydraulic power is available for operation of the FCS is limited to the last five minutes before launch; therefore, in order to have a reasonable margin of safety, the checkout time should not exceed two minutes.

- Reliability -- The reliability of the checkout equipment shall be a major design consideration. The airborne checkout is a final go/no-go test; and any malfunction indication, regardless of the cause, will probably result in aborting the flight.

- Display of Checkout Results -- All functions of the FCS shall be checked and indications given as to whether or not the system is operating properly. The go or no-go status of each distinct function shall be displayed separately so that the over-all system status can be determined in case the system is operating at less than maximum capability. For example, a low-altitude mission could be carried out even though a failure in the reaction control system were detected.

- Type of Tests -- All dynamic tests shall be of the analog variety and shall provide quantitive, repeatable results.

DEVELOPMENT OF CHECKOUT PROCEDURE

The development of the airborne checkout procedure involved establishment of:

1. A method of test whereby a complete checkout is made with a minimum number of tests.

2. A method of checking the adaptive or gain-changing features of the system.

3. The method of automating the equipment.

4. The detail test procedure and hardware requirements.

The problems encountered and the alternative solutions which were investigated during the development are described in the following paragraphs.

Test Method

The time limit on the checkout dictates that the number of tests be kept to a minimum. The most attractive scheme for minimizing the number of tests was to check the operation of the system on a functional basis from sensor input to servo output. Since trouble shooting and fault isolation (beyond functions) are not checkout requirements, there seemed to be little justification for breaking the system into individual circuits and components. A complet check of the system must include checkout of the FCS sensors (gyro, accelerometer, control stick, platform, etc.). Self-test gyros and accelerometers were requested which can be checked with an electrical torquing signal. Provisions were incorporated in the system to check for excitation and circuit continuity of the stick, platform, and Q-ball inputs. With these provisions, system operation can be checked by operating each sensor input circuit and checking the system output.

Output Signal Evaluation

After the general test method has been established, the next requirement is for a means of determining a go or no-go evaluation of the output signal. Two methods of doing this were investigated. The first was the conventional test technique using a voltage comparator and a reference voltage source. The second was a new method using a current summing comparator in a closed-loop test circuit. The new method makes use of the model-following characteristic of the adaptive flight control system. It has several advantages over the conventional method, the most significant being that it requires less than one-half the number of tests required by the conventional method. This is an important advantage because the number of tests required to check the fixed-gain and variable-gain systems, both of which are redundant, could not be run in the time allotted if the conventional technique were used. A detailed discussion of both methods follows.

Conventional Checkout -- The procedure used in conventional checkout of an adaptive control system is to apply known input signals to the system and compare the output voltage with a reference voltage. For example, the servo output for a control stick input would be checked as shown in Figure 4.

In addition to simulating the aircraft pitch rate response, the test circuit performs four functions for each comparator measurement:

1. It grounds all sensor inputs to the circuit.
2. It applies a simulated control stick signal to the system.
3. It connects the comparator to the servo.
4. It connects the comparator to the reference voltage source.

The aircraft simulation is necessary to establish the system gain. The voltage comparator operates a relay when one input voltage is larger than the other. Thus, two measurements are required to determine if a signal voltage is within its maximum and minimum limits: The first checks that the signal is greater than the lower limit; the second checks that the signal is less than the upper limit. The signal and reference inputs are reversed for the second measurement. Thus, six test configurations are required to check for null, positive, and negative signals.

The advantages of this method of testing are:

1. The system can be checked once while engaged to check the operation of the lead amplifier, servo amplifier, and servos.

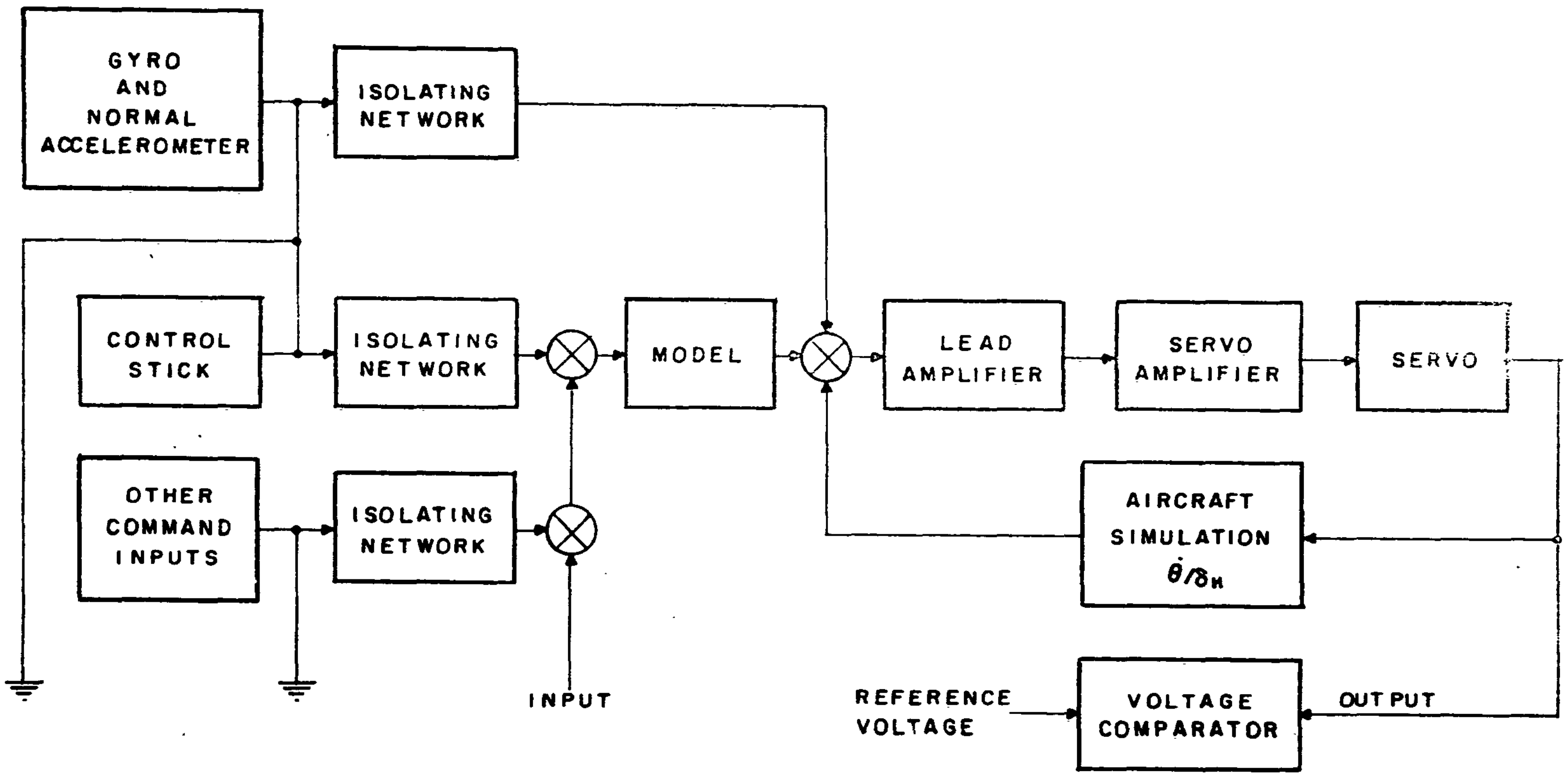


Figure 4. Conventional Method of Closed-Loop Testing

Then the system can be disengaged, the sensor signals applied, and the output of the model checked against the reference voltages. Having the system disengaged for the major portion of the checkout may be an important consideration during the airborne checkout, where surface motion may be undesirable.

2. The technique lends itself to trouble shooting because, by increasing the number of tests, the system can be broken up into individual amplifier cards and/or components.

3. The voltage comparator is a high input impedance device and will not load the circuit being tested.

Some disadvantages are:

1. The number of separate tests required (6 per input) results in quite complicated switching and relay logic.

2. The accuracy of the input signals and reference voltages must be closely controlled.

3. The high input impedance of the comparator requires shielding to prevent pickup and noise.

4. The procedure is practical only for steady-state voltages. Circuits containing integrators, for example, cannot be checked easily, and transients cannot be evaluated.

New Method -- The procedure used in the new method is to compare the aircraft simulated response to a command input directly with the output of the model, which is the desired response to the command. The comparator is a current-summing switching amplifier which is connected to operate a relay when the difference between the input signals exceeds a preset value. The test circuit for a control stick input check is shown in Figure 5.

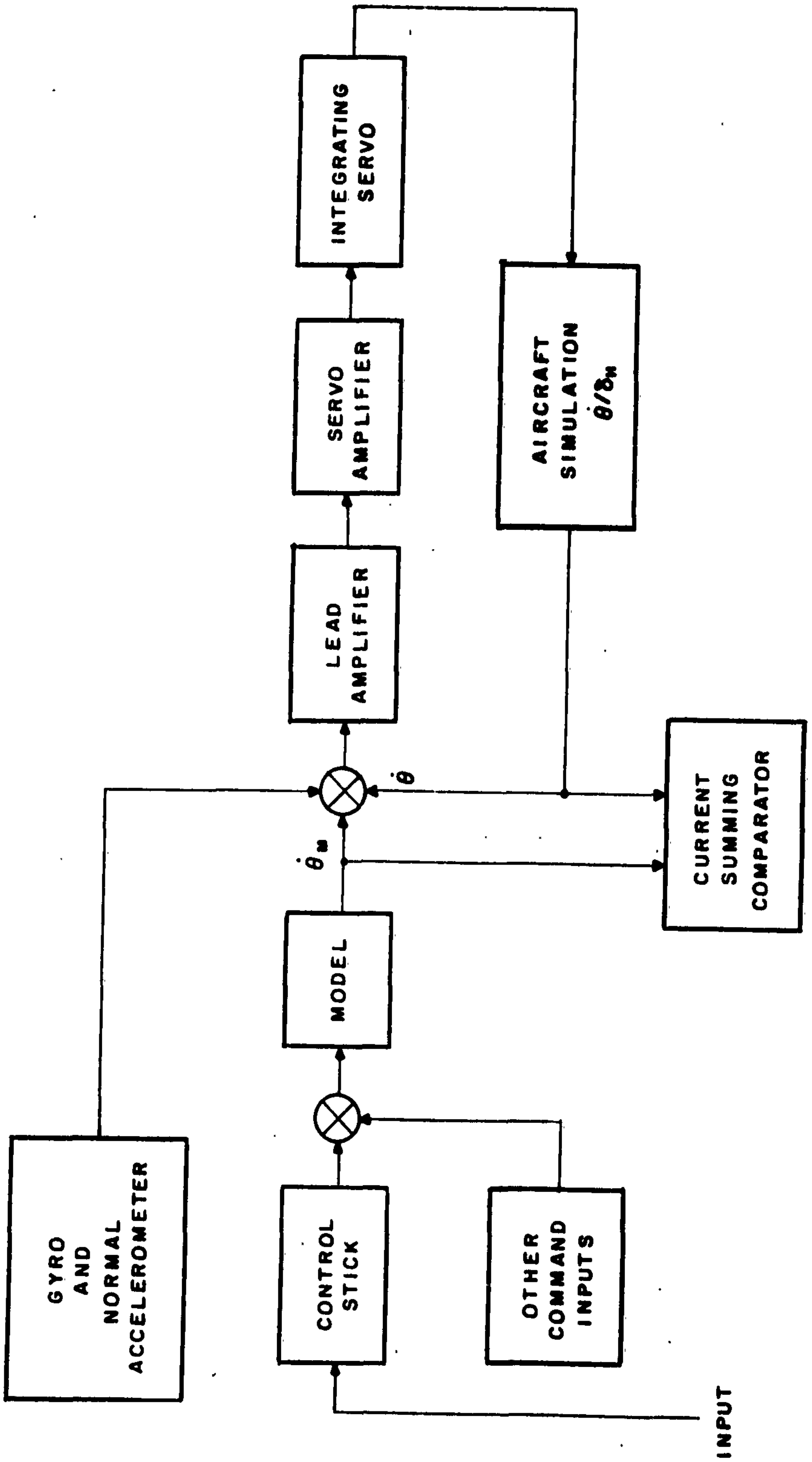


Figure 5. A New Method of Closed-Loop Testing

In this sytem the simulated aircraft pitch ratc signal replaces the reference voltage used in the conventional method. The input to the system in this case is introduced by shorting an end resistor on the stick potentiometer. The magnitude of the resulting signal is not critical because the aircraft response should follow the model for all command inputs. For the samé reason, the other inputs to the model do not have to be grounded.

All command inputs can be checked without switching the comparator leads. The gyro and normal accelerometer inputs are checked by comparing their outputs with the simulated aircraft response. In the case of the gyro, for example, this is done by switching one of the tester comparator inputs from the model to the gyro output. The gyro is operated by applying a current to the torquing winding through a lag network which has a response similar to that of the model. The simulated pitch rate will now follow the gyro input just as though it were a command input.

Some of the advantages of this closed-loop checkout procedure are:

1. The switching necessary for a system checkout is reduced by at least half.

2. The comparator is capable of performing a continuous check of both transient and steady-state system operation.

3. No regulated reference voltage sources are required.

4. The low-impedance circuitry is not susceptible to noise and pickup.

5. Fewer test points are required because no isolation is required where sensor signals must be grounded and test signals inserted.

It should be evident that the closed-loop technique provides a method of performing a complete operational test with a minimum of checkout equipment.

The disadvantages of this type of checkout procedure do not present any serious problems concerning the requirements for the airborne checkout. Some of the disadvantages are:

1. The current comparator loads the test circuit and can only be connected to points in the system which can supply the required current.

2. The system must be engaged for all tests. This means that the servos and surfaces will move during the checkout, though not extensively.

The closed-loop technique provides a check of over-all system performance rather than giving specific trouble shooting information. This is not a disadvantage, however, since specific trouble shooting information as to the particular component that has malfunctioned is of no value once the B-52 is airborne with the X-15. At this time only mode status is of value to the X-15 pilot and the MH-96 launch control operator.

Gain Computer Checkout

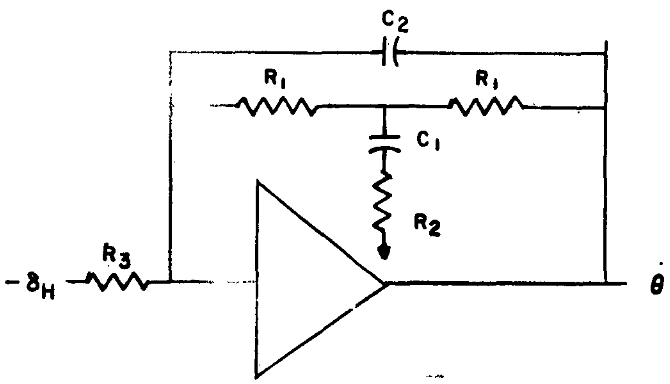
In order to perform operational tests on an adaptive flight control system, it is necessary to simulate the aircraft pitch, roll, and yaw rate response to surface deflection. This simulation is required because the gain of the system is automatically adjusted to a value inversely proportional to the effectiveness of the control surfaces. The gyro signal resulting from suface motion is the parameter used to indicate surface effectiveness and thus establish the system gain. An analog computer is used during the ground checks to computer these rate signals along with the other system parameters (angle of attack, attitude, acceleration, etc.). It is not feasible or necessary to duplicate this complete simulation in the airborne analyzer. Thus the problem becomes one of finding a less complex simulation which will give the required rate signals and approximate the transient response of the aircraft closely enough to determine whether the system is operating properly.

The possibility of using passive R-C networks to generate the 90-degree phase shift of the aircraft was investigated first. This simulation is the simplest and will establish a known system gain. However, a network of resistance and capacitance cannot simulate a second-order oscillatory aircraft and thus would not be an accurate simulation of transient conditions. Another disadvantage is that the current which can be drawn from such a network is limited by the size of capacitors available.

The use of active components appeared to be the best solution to the problem. Several methods were investigated in an effort to keep the number of operational amplifiers to a minimum. The pitch rate response to the elevator is a second-order equation which would normally require two integrators to solve:

$$\frac{\dot{\theta}}{-\delta_H} = \frac{-M_{\delta_H}}{T_a} \frac{1 + T_a S}{S^2 + 2 \zeta_a \omega_a S + \omega_a^2}$$

For flight conditions where T_a is a low value, the above equation can be solved with the following simulation using only one operational amplifier:



At flight condition No. 27

h	= 40,000 feet	$-M_{\delta_H}$	= 14.25 rad/sec^2
Mach	= 1.2	ω_a	= 3.8553 rad/sec
α_o	= 2 degrees	ζ_a	= 0.1936
q	= 395 lb/ft^2	T_a	= 1.636 second

The R and C values are:

R_1 = 14.6 kilohms C_1 = 1.4 microfarad

R_2 = 0.281 kilohms C_2 = 20 microfarads

R_3 = 50 kilohms

This simulation will provide the steady-state gyro signals to establish the gain level (approximately 25 per cent of maximum gain) as well as the second-order underdamped transient response. It is also versatile in that actuator lag can be simulated in the input and gyro lead network can be included in the output if the checkout requires this mechanization.

The problem of simulating the roll rate and yaw rate signals is more complicated because three simultaneous equations are involved:

Roll Rate

$$\dot{p} = L_p' p + L_\beta' \beta + L_r' r + L'_{\delta_r} \delta_r + L'_{\delta_a} \delta_a$$

Yaw Rate

$$\dot{r} = N_r' r + N_\beta' \beta + N_p' p + N_{\delta_r} \delta_r + N_{\delta_a} \delta_a$$

Sideslip

$$\dot{\beta} = Y_u \beta - r + \frac{g}{u_1} \phi + \frac{Y_{\delta_r}}{u_1} \delta_r + \frac{Y_{\delta_a}}{u_1} \delta_a$$

These equations evaluated at flight condition No. 27 become

$$\dot{p} = 1.6p - 6.48\beta - 0.12r + 16.3\delta_r - 47.3\delta_a$$

$$\dot{r} = -0.4r + 15.4\beta + 0.03p - 8.3\delta_r + 0.35\delta_a$$

$$\dot{\beta} = -0.26\beta - r + 0.024\phi + 0.088\delta_r + 0.0135\delta_a$$

Thus it appears that the $\dot{\beta}$ equations can be closely approximated by a first-order lag in r if terms containing the smaller coefficients are neglected.

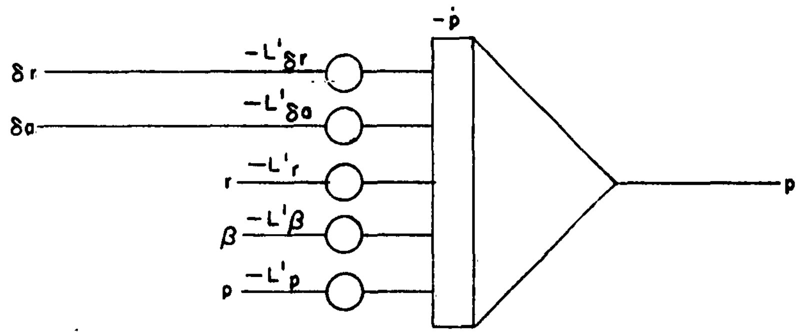
$$\dot{\beta} + 0.26\beta = -r$$

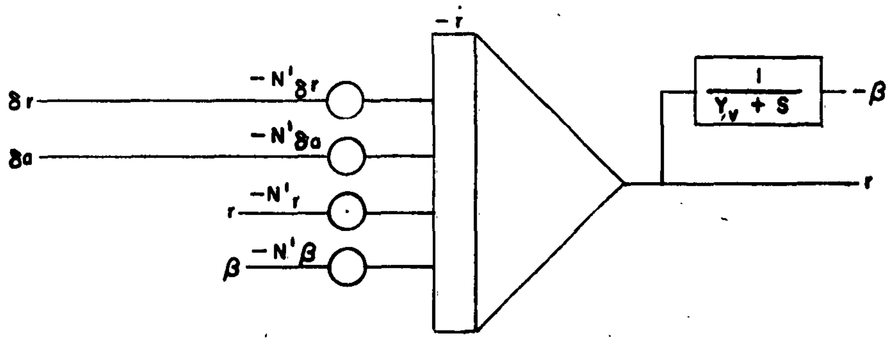
or

$$\dot{\beta} + Y_v\beta = -r$$

$$\beta = \frac{-r}{Y_v + S}$$

Now the two remaining equations can be solved simultaneously using two integrators and a lag to compute β from -r. Neglecting the 0.03p term in the $\dot{r}$ equation and changing signs, the simulation is





The above pitch and lateral simulations are shown connected to the redundant flight control system in Figure 6.

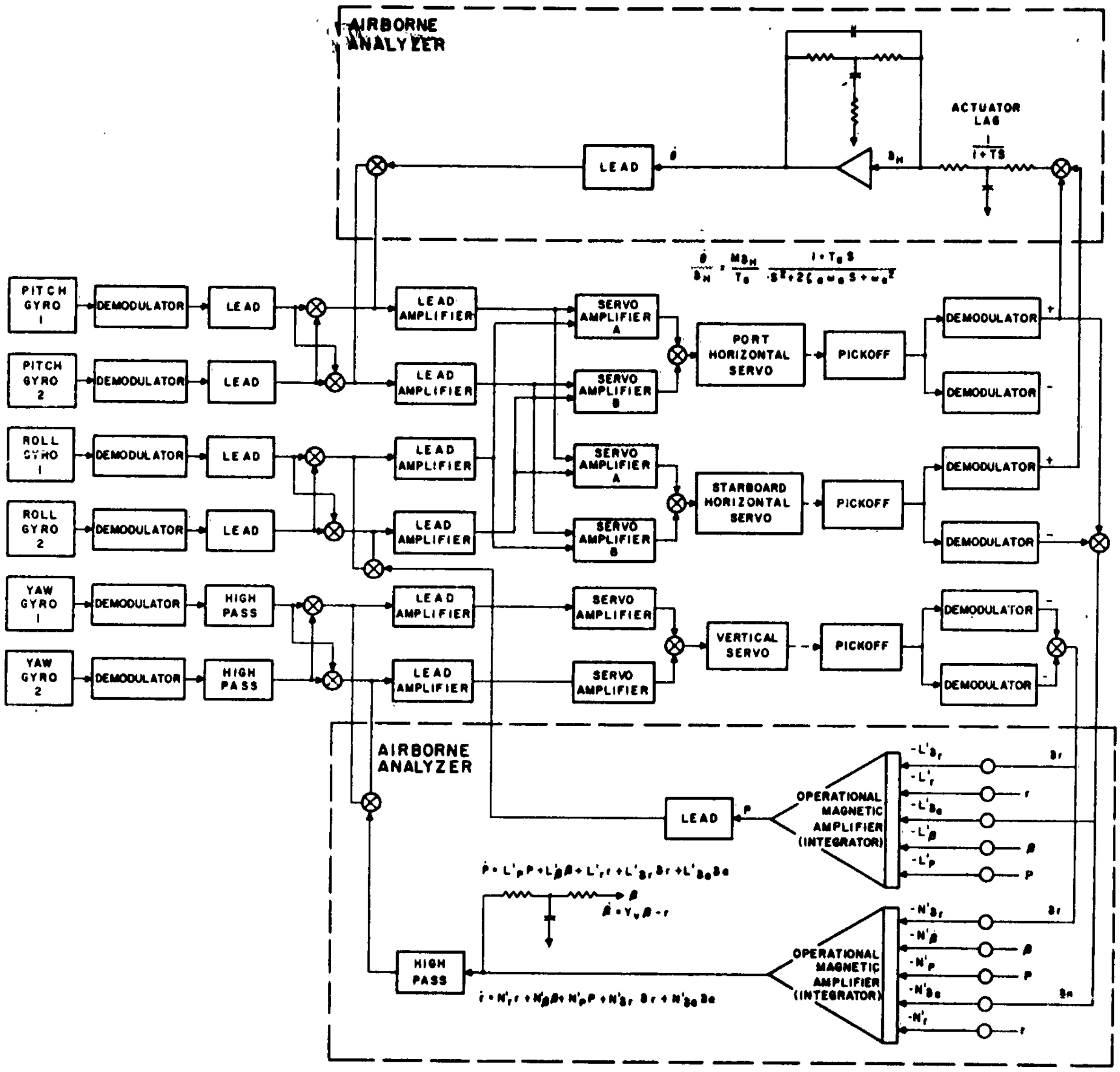


Figure 6. DUG1331 Airborne Analyzer Outer Loop Simulation Redundant Channels

Automatic Test Development

As the first FCS redundant block diagrams were received, it became apparent that the number of tests required to check the system would be two or three times the number required for a conventional system. It was therefore decided to make the airborne analyzer a completely automatic tester -- a decision which became a necessity as system complexity increased with the addition of redundant fixed-gain channels feeding both aero and reaction control systems. Approximately 50 input-output tests are required for the checkout. Two steps are used per test -- one with the input signal applied and one with it removed (for a null check); therefore, the total test capacity required is 100. Since, in general, three functions are switched per test (one system input signal lead and two comparator leads), the total switch capacity required is 300 switching circuits.

Telephone -type switches were selected to perform the basic switching functions. These switches offer the greatest capacity in the smallest package. They use double-wiping-action contacts and have life ratings of several million cycles. Although not normally used in airborne applications, enclosed versions are furnished which will meet the environmental requirements specified later in this section. The switches can be pulsed at 30 steps per second, a rate which greatly exceeds the airborne analyzer requirement of approximately one step per second.

The actual stepping from test to test is controlled by the comparator which operates a pulse generator. The output of the pulse generator advances the stepping switches at a uniform rate established by the time constant of the pulse generator. This mechanization prevents double pulsing or skipping.

The control unit is designed so that the automatic testing can be stopped at any time after initiation with either the STOP or RESET switch. It is automatically stopped when a malfunction is detected or at the end of the test. Once stopped, the automatic operation may be restarted by pushing the START switch. Manual operation is obtained by pushing the STOP switch and the the MANUAL ADVANCE switch. With this mechanization, the tester can be operated with a minimum of operator attention.

Detail Test Procedure

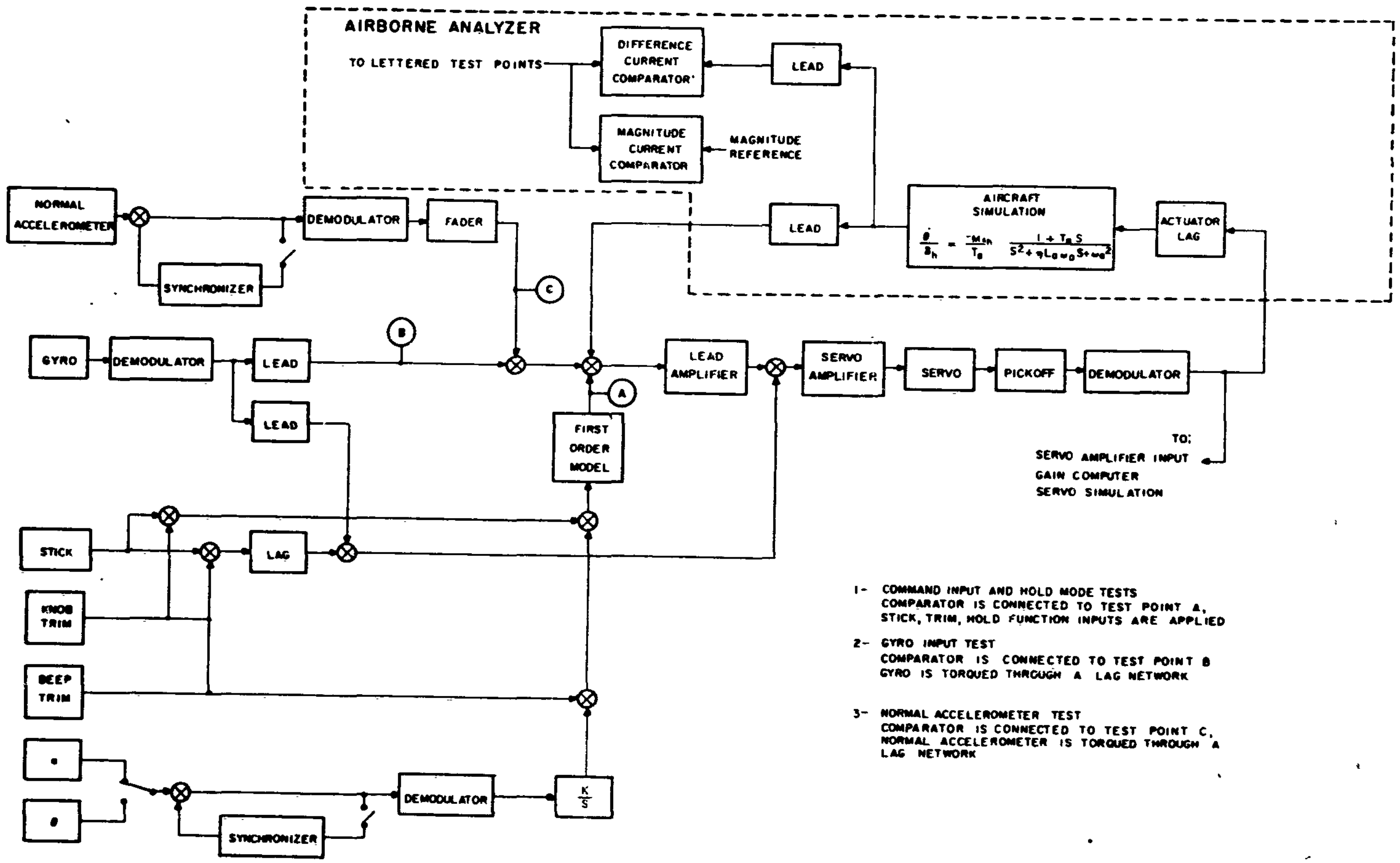
The following test procedure outlines the tests required for the complete FCS checkout. Figure 7 shows how the current comparators are connected into the system for three typical test situations. The function of the magnitude comparator is to verify that test signals are actually getting into the system. It is required because the difference comparator will show no difference in signals (a GO indication) when an input fails to cause the system to operate.

The complete checkout is divided into four groups of tests:

1. Pitch axis aero controls
2. Roll axis aero controls
3. Yaw axis aero controls
4. Reaction controls (all axes)

The types of tests run on the aero control functions in each axis include checks of fixed-gain functions, variable-gain functions, redundant variable-gain channel operation, failsafe functions, autopilot hold modes, and trim functions. The analyzer automatically operates the FCS engage circuitry, supplies the test signals, and checks the output response for each FCS input. The reaction control system is checked by determining that 28-volts direct current is present at the outputs of the reaction control switching amplifiers.

The test procedure is given in outline form in Table 4.



1- COMMAND INPUT AND HOLD MODE TESTS
COMPARATOR IS CONNECTED TO TEST POINT A,
STICK, TRIM, HOLD FUNCTION INPUTS ARE APPLIED

2- GYRO INPUT TEST
COMPARATOR IS CONNECTED TO TEST POINT B
GYRO IS TORQUED THROUGH A LAG NETWORK

3- NORMAL ACCELEROMETER TEST
COMPARATOR IS CONNECTED TO TEST POINT C,
NORMAL ACCELEROMETER IS TORQUED THROUGH A
LAG NETWORK

Figure 7. DUG1331 Airborne Analyzer - Block Diagram Pitch Axis Checkout

Table 4. Test Procedure

PITCH AXIS TESTS

Fixed-Gain System

1. Operate lead amplifier comparator to disengage variable-gain system.

2. Engage pitch damper.

3. Connect comparator to minimum-gain lag network output. Check stick response, knob trim response, and beep trim response.

4. Connect comparator to gyro lead output. Check gyro response.

Variable-Gain System (Damper)

1. Remove signal from lead amplifier comparator to enage variable-gain system.

2. Connect comparator to model output. Check stick response, knob trim response, beep trim response, α hold response, and θ hold response.

3. Connect comparator to gyro lead output. Check gyro response.

4. Connect comparator to Nz fader output. Check normal accelerometer response.

Redundant Channel Tests

1. Drive No. 1 gain computer to minimum. Drive No. 2 gain computer to two times normal value.

2. Connect comparator to model output. Check response to stick (should be same as normal operation).

3. Drive No. 1 gain computer to two times normal value. Drive No. 2 gain changer to minimum.

4. Check response to stick as in (2).

5. Remove inputs to gain computer.

Table 4. (Cont)

Failsafe Tests

Stick:

1. Operate Nos. 1 and 2 stick disengage comparators.

2. Check model output for stick inputs (should be null).

Gyros:

1. Connect comparator to gyro lead outputs. Operate Nos. 1 and 2 gyro comparators.

2. Torque gyros and check for null output.

Normal Accelerometers:

1. Connect comparator to model output. Operate Nos. 1 and 2 normal accelerometer comparators.

2. Torque accelerometers. Check for null model output.

Lead Amplifier: Already checked during minimum-gain tests.

Servo Monitor:

1. Operate No. 1 servo monitor comparator (damper will disengage). Check null output of aircraft response for stick input.

2. Operate No. 2 servo monitor comparator. Again check null output of aircraft response to stick input.

NOTE: The above pitch axis checkout requires 20 comparator measurements. (Checkout is complete except for reaction control tests.)

ROLL AXIS TESTS

Fixed-Gain System

1. Operate lead amplifier comparator to disengage variable-gain system.

2. Connect comparator to minimum-gain model output. Check response to lateral stick input.

3. Connect comparator to gyro lead output. Check response to gyro input.

Table 4. (Cont)

Variable-Gain System

1. Remove signal from lead amplifier comparator to engage variable-gain system.
2. Connect comparator to roll model output. Check responses to lateral stick input, roll trim input, roll attitude input, and heading attitude input.
3. Connect comparator to gyro lead output. Check response to gyro input.

Redundant Channel Tests

1. Drive No. 1 gain computer to minimum. Drive No. 2 gain computer to two times normal value.
2. Connect comparator to model output. Check response to lateral stick input (should be same as normal operation).
3. Drive No. 1 gain changer to two times normal value. Drive No. 2 gain changer to minimum.
4. Check response to stick input as in (2).

Failsafe Tests

Stick:

1. Operate Nos. 1 and 2 stick disengage comparators.
2. Check model output for stick input (should be null).

Gyros:

1. Operate Nos. 1 and 2 gyro disengage comparators.
2. Connect comparator to gyro lead output. Check gyro null output for gyro torquing signal.

Lead Amplifier: Already checked during minimum-gain tests.

NOTE: Roll checkout contains 12 comparator measurements.

Table 4. (Cont)

YAW AXIS TESTS

Fixed-Gain System Tests

1. Operate lead amplifier comparator to disengage variable-gain system. Engage roll damper.

2. Connect comparator to fixed-gain lead + high-pass output. Check response to yaw gyro input.

3. Connect comparator to attenuator network output. Check response to lateral accelerometer input.

Variable-Gain System Tests

1. Remove signal from lead amplifier comparator to engage variable-gain system.

2. Connect comparator to output of high-pass networks. Check response to gyro input.

3. Connect comparator to lag network output. Check response to lateral accelerometer inputs.

Redundant Channel Tests

1. Drive No. 1 gain computer to minimum. Drive No. 2 gain computer to two times normal value.

2. Check response to lateral acceleration input (should be same as normal response).

3. Drive No. 1 gain computer to two times normal value. Drive No. 2 gain computer to minimum.

4. Check response to lateral accelerometer as in (2).

Failsafe Tests

Gyros:

1. Connect comparator to gyro high-pass outputs. Operate Nos. 1 and 2 gyro disengage comparators.

2. Check null output of gyros for torquing signal.

Table 4. (Cont)

Rudder Pedals:

1. Connect comparator to rudder pedal lag output. Operate Nos. 1 and 2 rudder pedal disengage comparators.

2. Check null output for pedal input.

Lead Amplifier:

1. Apply signal to No. 1 servo monitor (yaw damper will engage).

2. Connect comparator to gyro simulation. Check null output of aircraft response to gyro inputs.

3. Apply signal to No. 2 servo monitor.

4. Check null output of aircraft response to gyro inputs.

NOTE: The above yaw axis checkout requires 10 comparator measurements.

REACTION CONTROL TESTS

1. Disengage aero control servos.

2. Drive gain to 80 per cent of maximum in all three axes.

3. Connect comparator to appropriate output of reaction control switching amplifier. Check reaction control outputs for pitch-up stick, pitch-down stick, left roll stick, right roll stick, left rudder, right rudder, left heading error, right heading error, plus yaw rate input, and minus yaw rate input.

NOTE: Reaction control checkout requires 10 comparator measurements.

TOTAL COMPARATOR TESTS

The total number of comparator tests to be run for the entire checkout is:

Pitch	20
Roll	12
Yaw	10
Reaction Control	10
	52

ACCELEROMETER TESTING BOOM

Although the normal and lateral accelerometers used by the MH-96 FCS contain self-testing features, it is still necessary periodically to remove them from the aircraft for performance tests. These tests consist of linearity, hysteresis, accuracy, cross-coupling, and threshold checks. They are most easily accomplished by using a special testing fixture to mount the accelerometers in various positions at a known radius from the center of a rate-of-turn table. Spinning the table at a fixed rate will produce precise accelerations upon the devices.

The existing SAS rate table was found to be unacceptable at low turning rates because of erratic outputs. Therefore, the NASA, Edwards AFB rate table (Genesco Model C-181) was used as a standard; and the accelerometer testing boom was designed to fit the mounting holes of this unit. A more detailed description of this device is found in Section IV.

REDUNDANCY AND FAILSAFETY

The MH-96 FCS is a dual redundant system in all three axes. In addition, it has both a variable-gain and a fixed-gain signal path within each of the six channels. Thus, signal levels at any one point within the adaptive calibrator are a combination of many inputs. The levels of these inputs depend in most cases upon the flight condition being simulated and upon the engaged or disengaged status of the various channels and of the sensor disengage monitors. The redundant mechanization within the adaptive calibrator tends to mask failures and make the checkout procedures more difficult. As a result of this redundancy, the number of tests required to verify system performance is increased by a factor greater than the number of redundant channels within each axis. As an indication of the number of tests required to adequately check all the inputs to the four horizontal servo amplifiers, over 100 open-loop tests are necessary.

ADAPTIVE CHANNEL MONITORS

The variable-gain amplifiers shown in Figure 8 are monitored by the current-sensitive relay K_a and its associated zener diodes. Whenever the difference voltage output from the variable-gain amplifiers is of sufficient magnitude, relay K_a is energized and its contacts remove both variable (or adaptive) gain channels in a given axis. This comparator is particularly difficult to check because this differential signal is normally zero and the inputs to each half of the redundant adaptive channel are identical.

In order to check the adaptive channels separately, it was necessary to incorporate within the variable-gain amplifiers a method of applying inputs to each channel directly rather than applying signals to the normal signal inputs. The relay K_a may be checked for normal operation by applying opposite-polarity d-c signals simulating a channel hardover output condition to cause the relay to operate.

The test points within the gain computers may be used during open-loop tests to set the gain of any channel at a known value. Thereafter, the channel may be tested as a linear system; that is, specific test points now would have predictable values of measured voltage corresponding to those of a known good system. To accomplish this constant gain condition within the variable-gain channels, it is necessary only to ground a point within the gain computers (not shown in Figure 8). This grounding connection drives all variable gains to their minimum value or, in effect, to the gains that would be present when the adaptive system was first engaged and the gains had not yet had time to rise to their correct values. With the gains set at minimum value, the insertion of a calibrated current into any gain computer will cause the gain to rise to a constant, known value. The normal fixed-gain channels which parallel the adaptive or variable-gain channels are still present to cause surface movement. If further isolation is desired, the only way to separate the fixed-gain inputs to the servo amplifiers from the minimum gain adaptive inputs is to trigger relay K_a as explained above.

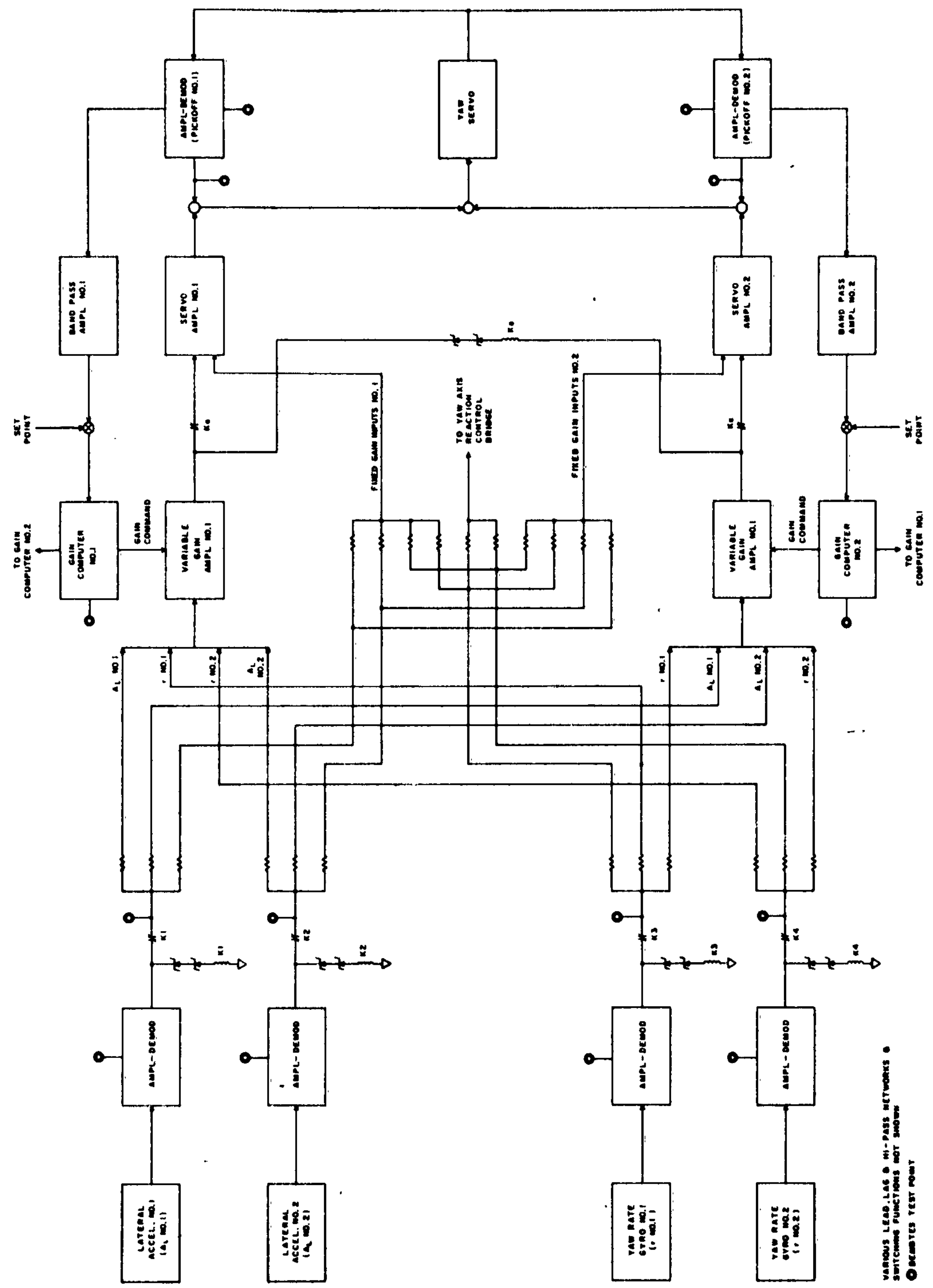


Figure 8. Simplified MH-96 FCS Yaw Axis

COMPARATORS

The MH-96 FCS uses comparators on almost all inputs to the adaptive calibrator with the exception of such inputs as the angle-of-attack error signal or pitch and roll attitude error signals. These comparators are set to trip whenever any sensor output, after demodulation, reaches a preset d-c level; a protection against hardover failures within the sensors or the amplifier demodulators is thus provided. Other comparators are employed to disengage the adaptive-gain channels, but not the fixed-gain channels, whenever there is a disagreement between the two adaptive gains in each axis. A simplified yaw axis diagram is shown in Figure 8, and this drawing will be used to illustrate the methods by which these failsafe comparators may be activated or checked for isolating redundant channel malfunctions.

The method for checking the tripping level of the sensor input monitors or of removing a sensor input during a channel test is very similar to the method that causes the monitor to trip in actual practice. For example, if it were desired to test the tripping level of the yaw rate gyro No. 1 comparator, a calibrated a-c current would be inserted into the test point attached to the yaw rate gyro No. 1 amplifier-demodulator. This current causes a corresponding voltage to appear at the output of the demodulator. The corresponding voltage should trip the comparator, causing relay coils K-3 to be energized when its associated zener diode breaks down. This, in turn, will remove the demodulator output voltage from the system when the contacts of K-3 open. A test point is provided beyond these contacts to determine when relay action occurs. Similar methods will be employed on all sensor comparators within the adaptive calibrator.

SELF-TEST PROCEDURES

By employing the circuitry shown in Figure 9, the accelerometers are tested while still installed in the X-15. These units are tested by applying a specified

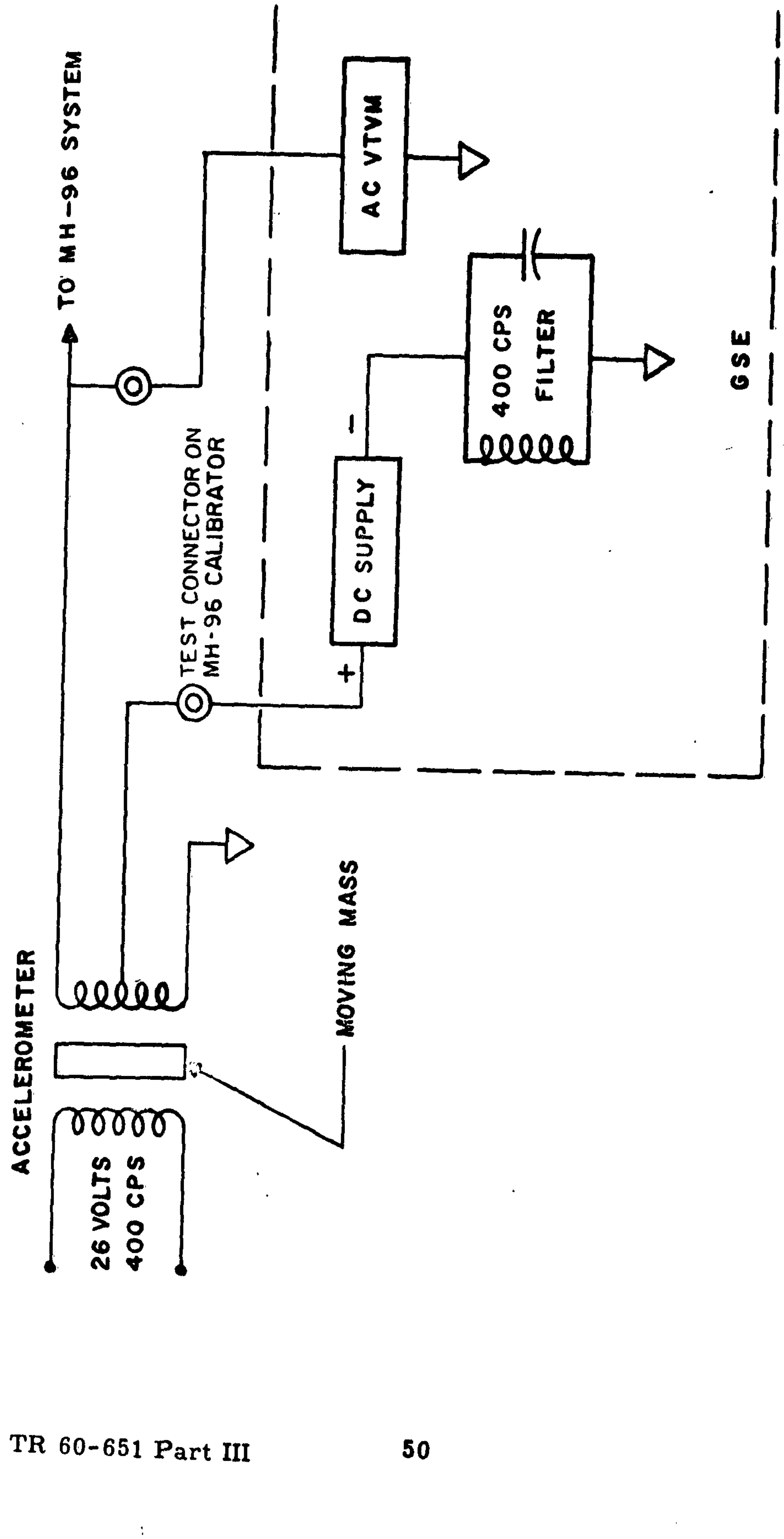


Figure 9. Accelerometer Self-Test

d-c voltage from the flight line analyzer to an unused center tap of each of the accelerometer output windings. This voltage causes solenoid-type action to be applied to the moving mass and results in the development of a coupling of the a-c primary voltage into the secondary or pickoff winding. This output voltage may be measured by the vacuum-tube voltmeter in the flight line analyzer and is a quantitative measurement of accelerometer performance. Both in-phase and out-of-phase signals may be obtained by rearranging the d-c connections and grounds.

The MH-96 rate gyros are tested while still installed in the X-15, as shown in Figure 10. A calibrated d-c current is applied to the unused pickoff windings of each rate gyro. The currents in the primary and secondary of this pickoff interact to develop a torque which displaces the gyro against its restraining springs. The displacement of the signal pickoff coil causes an output signal to be developed that is proportional to the current product of the d-c currents. Both in-phase and out-of-phase signals may be generated by varying current directions. The output voltage may be measured with the vacuum tube voltmeter that is contained in the flight line analyzer, and quantitative measurements of rate gyro performance may be obtained.

ENVIRONMENTAL AND LIFE TESTING

Because of the extensive environmental and life test program planned for the MH-96 FCS, a rapid method of evaluating system performance during the two-hour environmental cycles was required. The DUG1352 bench mockup is being designed to perform these tasks automatically with the additional capability of complete manual system checkout.

TEST MODES

The automatic mode will consist of a series of tests to be run automatically three times in each environmental cycle. The following ground rules were formulated to interpret failures detected by the automatic testing:

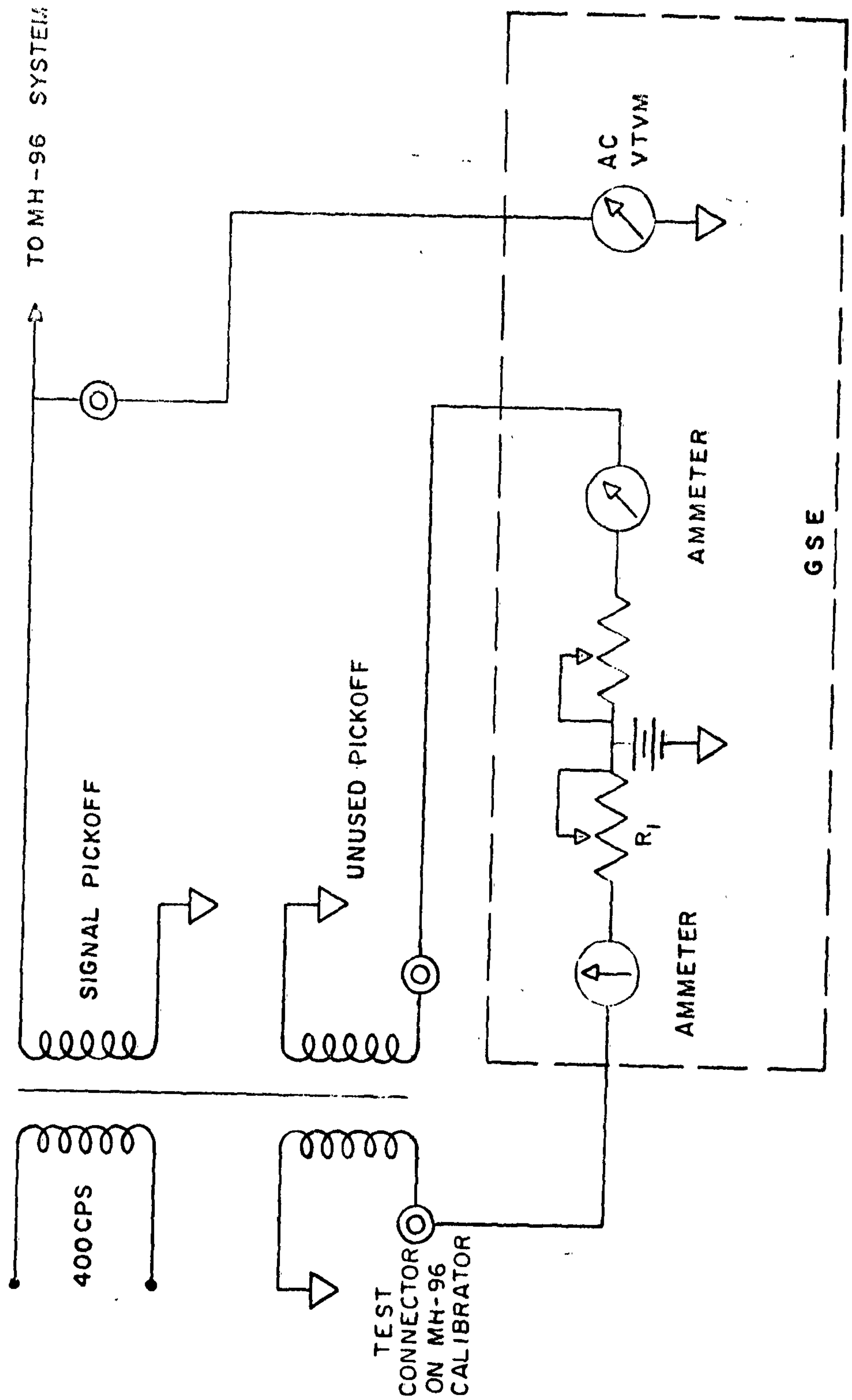


Figure 10. Rate Gyro Self-Test

1. The automatic mode will be run three times in each environmental cycle.

2. A program will be established for the automatic mode that will test all major components of the adaptive calibrator and perform a closed-loop test of all redundant channels. These tests will be determined by Honeywell and WADD to include as much testing as possible within the intended time and performance limits of the bench mockup.

3. If a malfunction occurs in a single channel of the adaptive system and does not result in a hardover output, the tests will be stopped, the malfunction repaired, and a complete manual engineering specification test run on the calibrator. The malfunction will not be charged to the system. Only those malfunctions resulting in the loss of the adaptive portion of the system or those resulting in a hardover output will be charged against the system.

4. A complete engineering specification test will be performed approximately every 25 cycles. At this time if the manual checkout reveals one of the very small number of malfunctions that might conceivably be missed by the automatic mode, it will be analyzed for the type of failure and then assumed to have occurred during the first of the 25 previously performed cycles.

The automatic tests will consist of the same series detailed in the previous discussion of the airborne analyzer. This similarity occurs because the two devices have essentially the same task to perform; i.e., to determine quickly the functional status of the MH-96 FCS. Neither device is required to trouble shoot the system. In the air no failures can be repaired, and during the environmental cycles no detailed trouble shooting can occur until the faulty major component of the MH-96 FCS has been removed from the environmental chamber. At this time the manual capabilities of the bench mockup, along with the services of a skilled technician, are available to give the component a thorough and comprehensive engineering specification test.

SERVO AND AIRCRAFT SIMULATION

The servo and aircraft simulation used in the manual and automatic modes of test is shown in Figure 11. This simulation is required to set the gain of the adaptive loops at a known chatter frequency and amplitude. Only the servo portion is required for use in the automatic mode, since another aircraft simulation is self-contained in the automatic mode portion of the device. Here a more elaborate simulation is used for aircraft response, which includes an oscillatory aircraft as explained in the discussion of the airborne analyzer. For simple loop closure in the manual mode, a $k/(1 + 0.33S)$ aircraft simulation is used as shown in Figure 11.

Two different kinds of feedback to the MH-96 system are required for closed-loop testing in either mode: One is the servo position feedback which is used to indicate the amplitude and frequency of the servo motions; the second is used, after appropriate lags have been added, to simulate aircraft rate response to surface motion. This information is fed back to the MH-96 rate gyro inputs and helps establish system gain. The K in the aircraft transfer function is a variable that will be used to vary system gain. Three axes are simulated simultaneously.

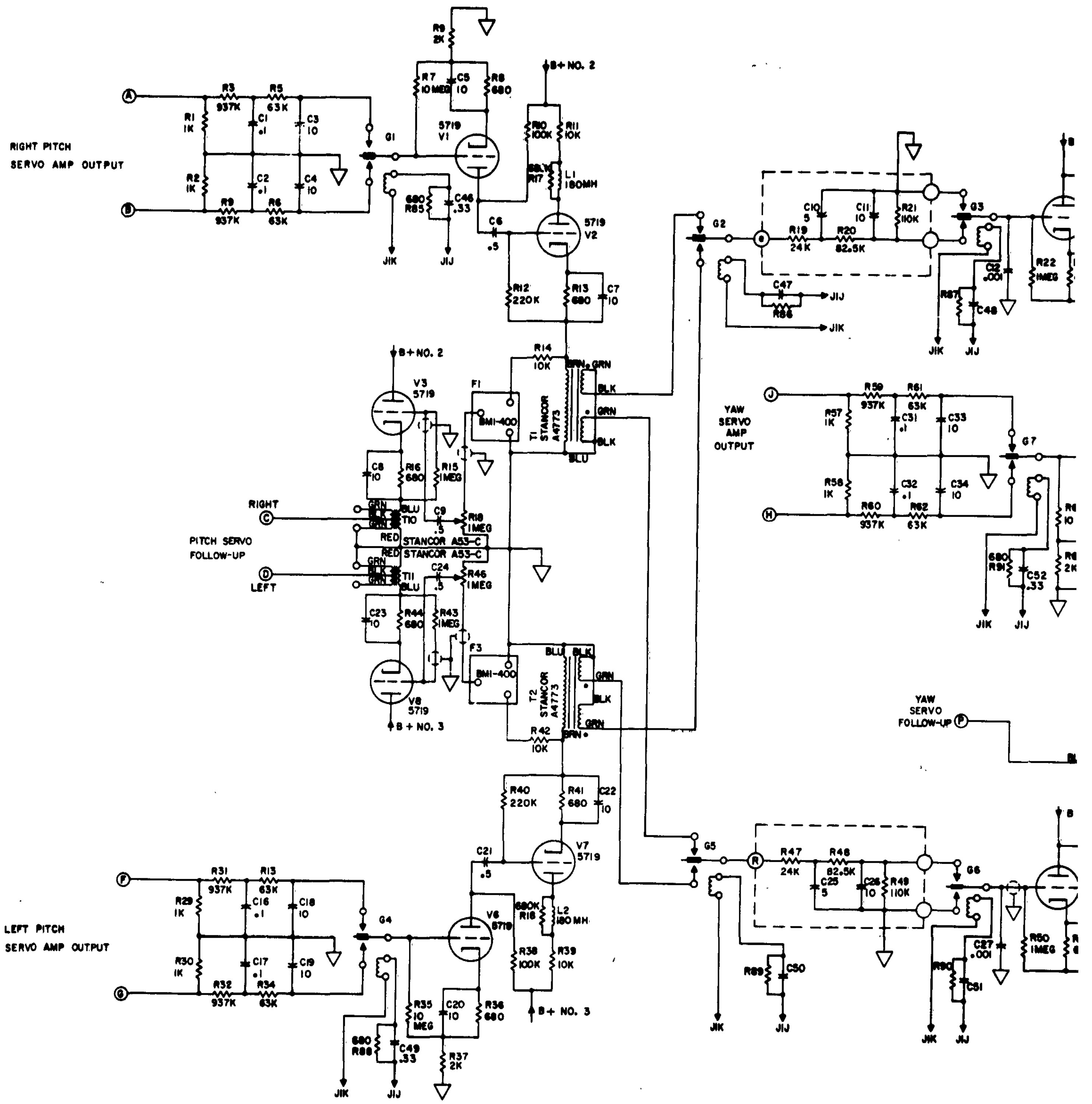


Figure 11. Servo and Aircraft

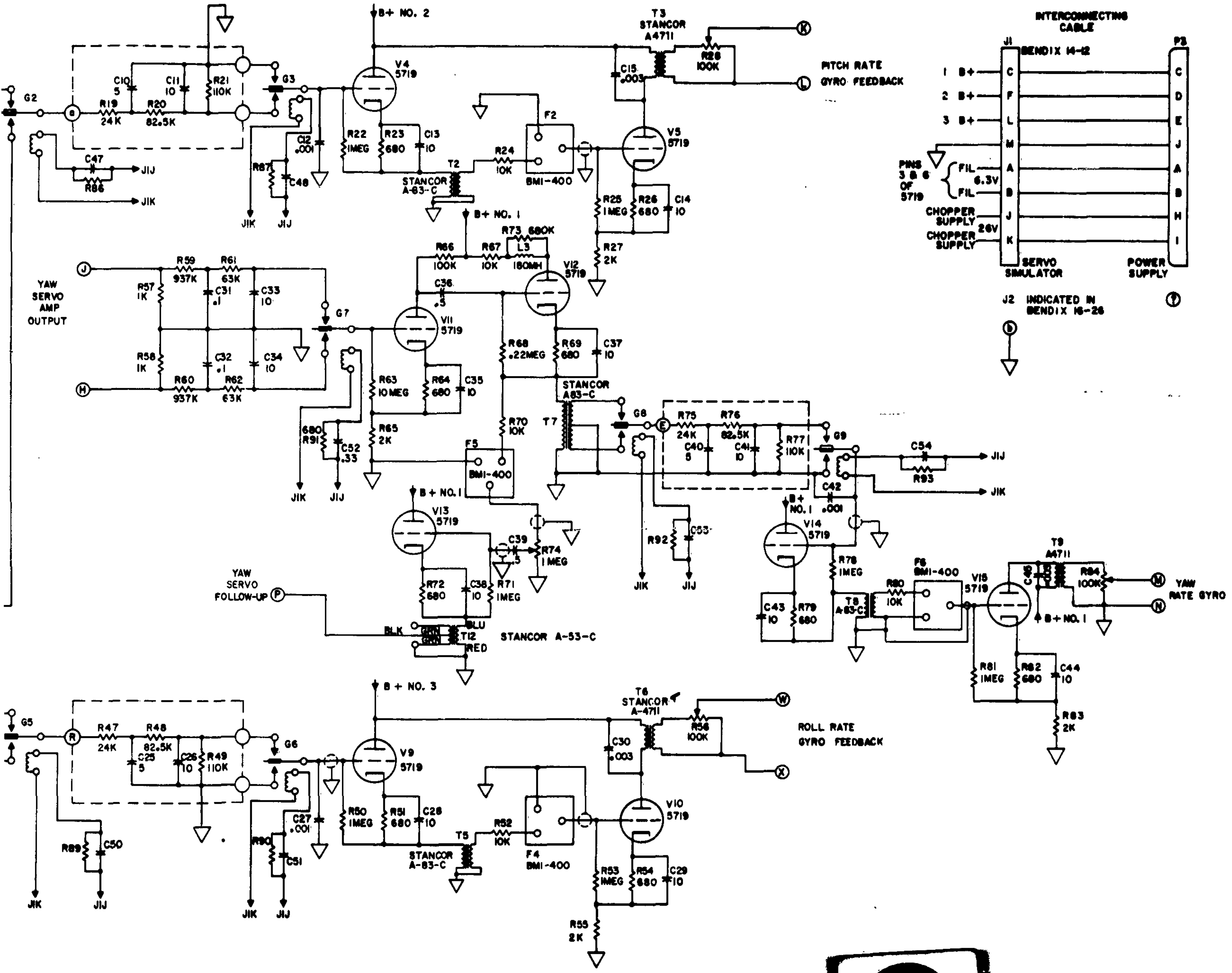


Figure 11. Servo and Aircraft Simulation

SECTION IV
EQUIPMENT DESCRIPTION

DUG1120A1 FLIGHT LINE ANALYZER

The DUG1120A1 Flight Line Analyzer, shown in Figures 12 and 13, is a mobile field level test set designed to be manually programed. The entire flight line analyzer has been constructed in accordance with Military Specification, MIL-T-21200, for ground support equipment. It includes a number of standard instruments such as an oscilloscope, an eight-channel oscillograph with associated preamplifiers, a low-frequency function generator, and a transistorized analog computer. These components are shown in relation to each other on the operator's console of the tester in Figure 13.

The input power to the analyzer is 115 volts, 60 cps; 115 volts, 400 cps; and 28 volts direct current. The four-wheeled mobile unit which houses the analyzer is 90 inches long, 58 inches high, and 48 inches wide. The flight line analyzer weighs approximately 1700 pounds.

Four bays are provided in the side of the analyzer opposite the operator's console. These bays mount the chassis which contain all the interconnecting circuits and components for controlling the various modes of testing and for preprograming the analog computer. Modular construction is used throughout the bays to facilitate trouble shooting and repair of the device. Storage drawers and bins are also provided in these bays to house all accessory gear and components associated with the standard instruments. Cables for connecting the flight line analyzer to the FCS are attached to the analyzer and stored in a covered bin located above these four rear bays.

The flight line analyzer is provided with a blower for forced-air cooling of all electronic components.

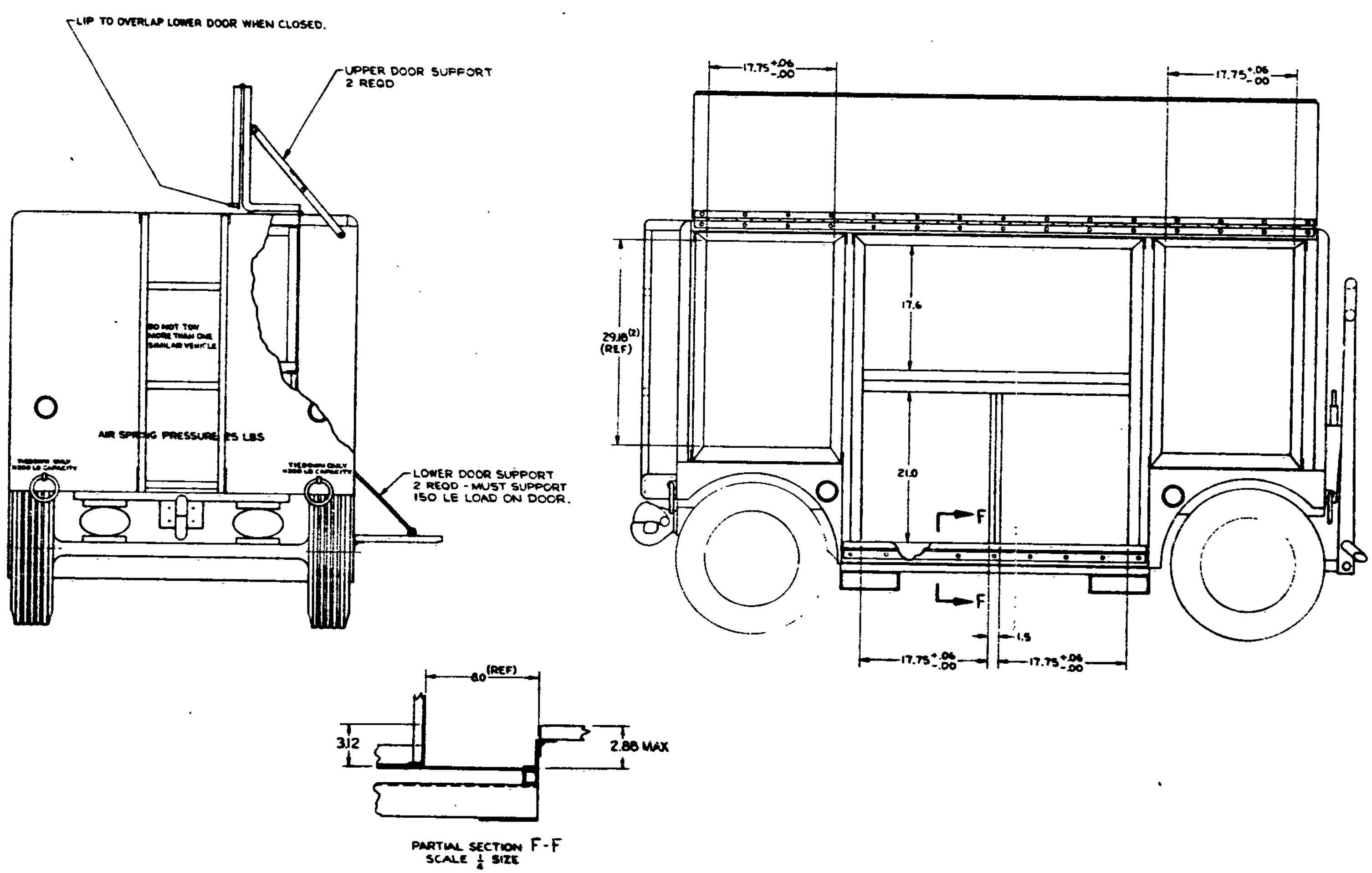
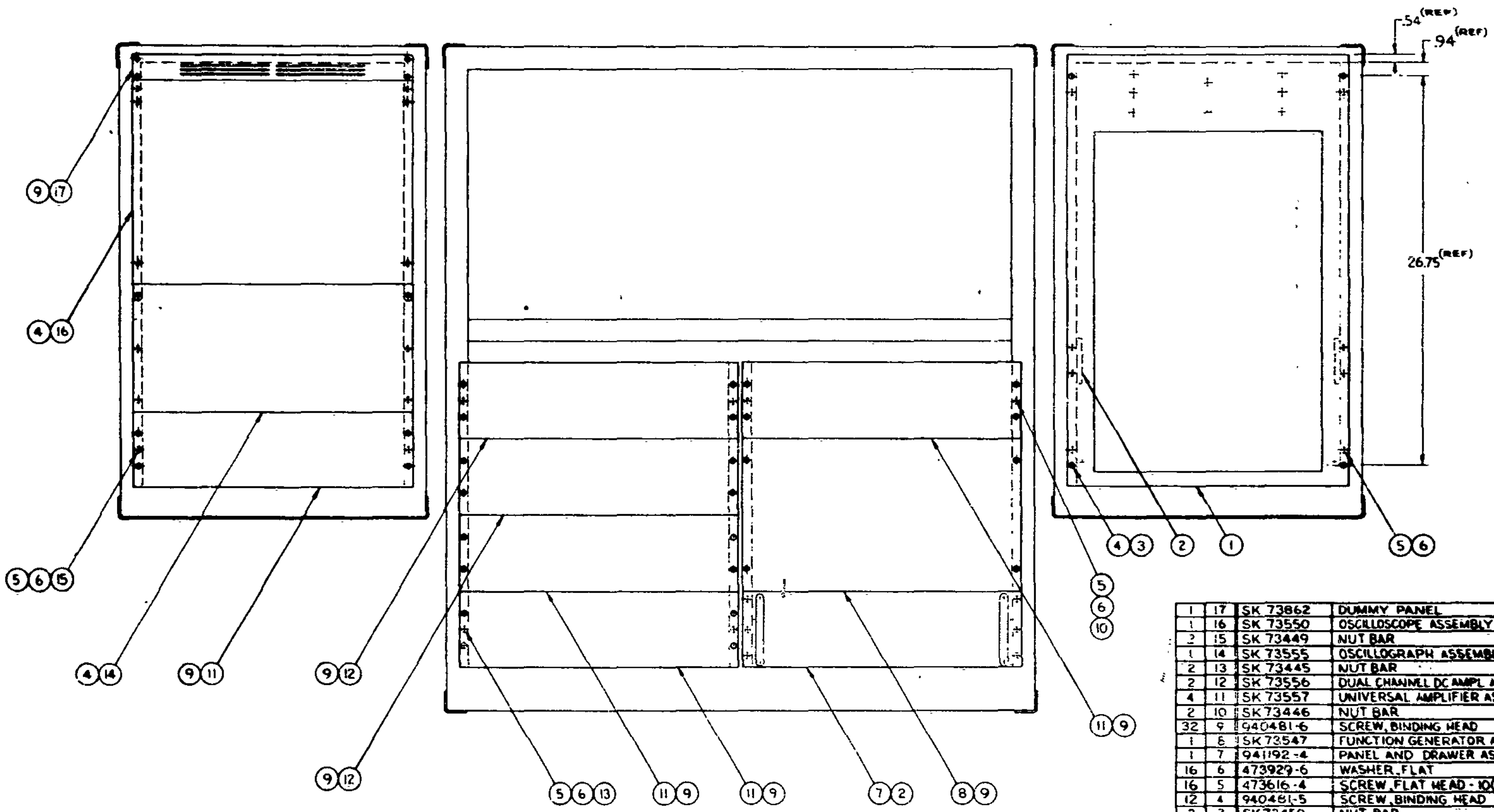


Figure 12. DUG1120 Flight Line Analyzer



QTY	ITEM	PART NO.	DESCRIPTION
1	17	SK 73862	DUMMY PANEL
1	16	SK 73550	OSCILLOSCOPE ASSEMBLY
2	15	SK 73449	NUT BAR
1	14	SK 73555	OSCILLOGRAPH ASSEMBLY
2	13	SK 73445	NUT BAR
2	12	SK 73556	DUAL CHANNEL DC AMPL ASSY
4	11	SK 73557	UNIVERSAL AMPLIFIER ASSY
2	10	SK 73446	NUT BAR
32	9	940481-6	SCREW, BINDING HEAD
1	8	SK 73547	FUNCTION GENERATOR ASSY
1	7	941192-4	PANEL AND DRAWER ASSY
16	6	473929-6	WASHER, FLAT
16	5	473616-4	SCREW, FLAT HEAD - 100°
12	4	940481-5	SCREW, BINDING HEAD
2	3	SK 73450	NUT BAR
2 PAIR	2	AP 941165-21	CHASSIS TRACK
1	1	SK 73443	COMPUTER PANEL ASSEMBLY

Figure 13. Layout of Flight Line Analyzer Modules

DUG1331 AIRBORNE ANALYZER

The DUG1331 Airborne Analyzer will be supplied as two units: the control panel and the switching unit.

Control Panel

The control panel outline drawing is shown in Figure 14. Green lights are used to show the operational status of the FCS. The malfunction lights are red and the test set status lights and the control push buttons are amber.

The physical specifications are:

Size - 12 x 8 x 4 inches

Weight - 5 pounds maximum

Mounting - Vertical panel mounted

Switching Unit

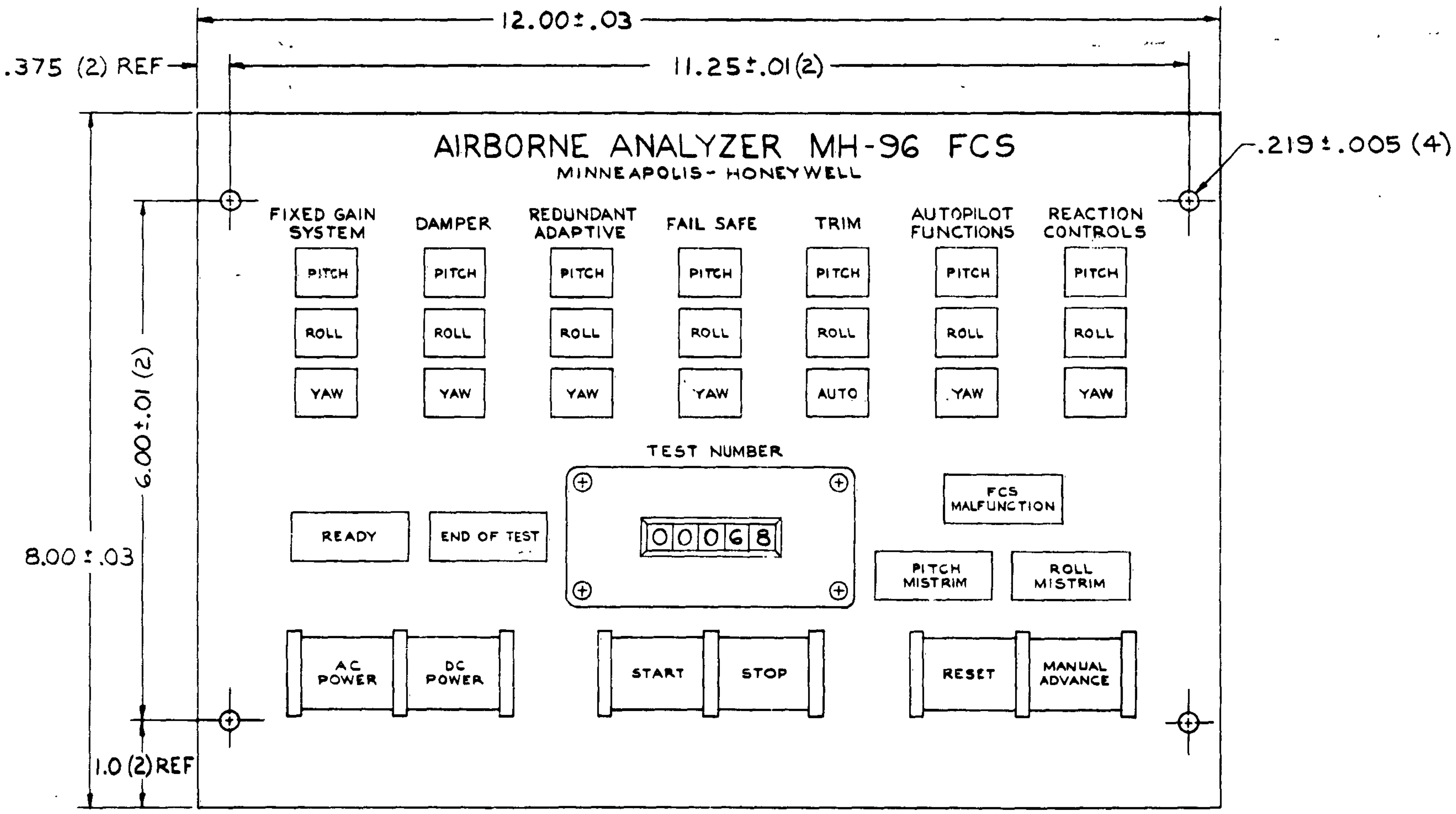
The switching unit outline drawing is shown in Figure 15. This unit contains the stepping switches, relays, comparators, signal sources, etc. The power circuit breakers and calibrations adjustments are located on the front panel. Test points for both the system and test set are provided to aid in trouble shooting the equipment.

The physical specifications are:

Size - 15.375 x 7.625 x 19.562 inches

Weight - 75 pounds maximum

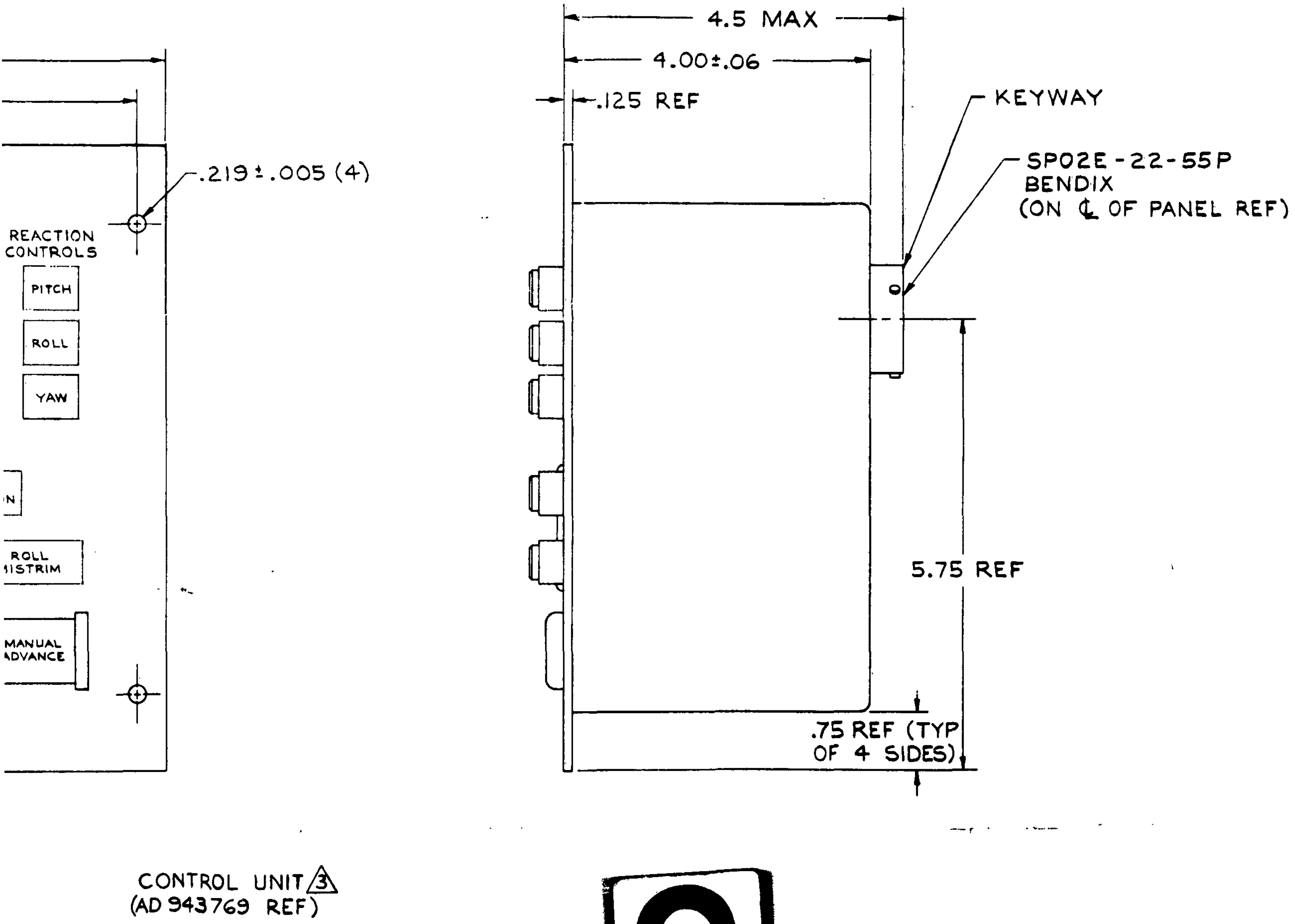
Mounting - MS91405 - CID Base



CONTROL UNIT 3
(AD 943769 REF)

1

Figure 14. DUG1331 Operator's Cont



2

e 14. DUG1331 Operator's Control Unit

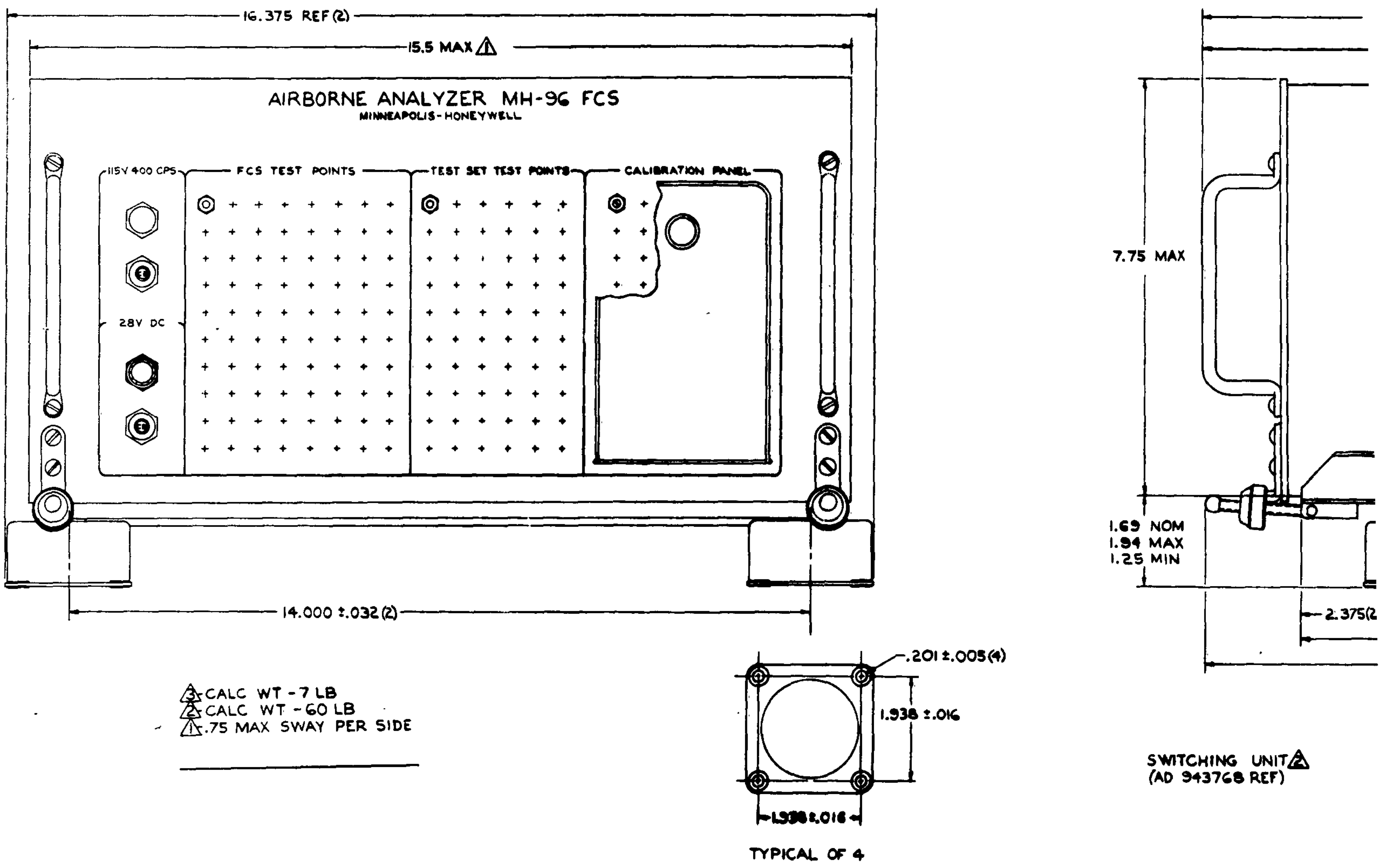
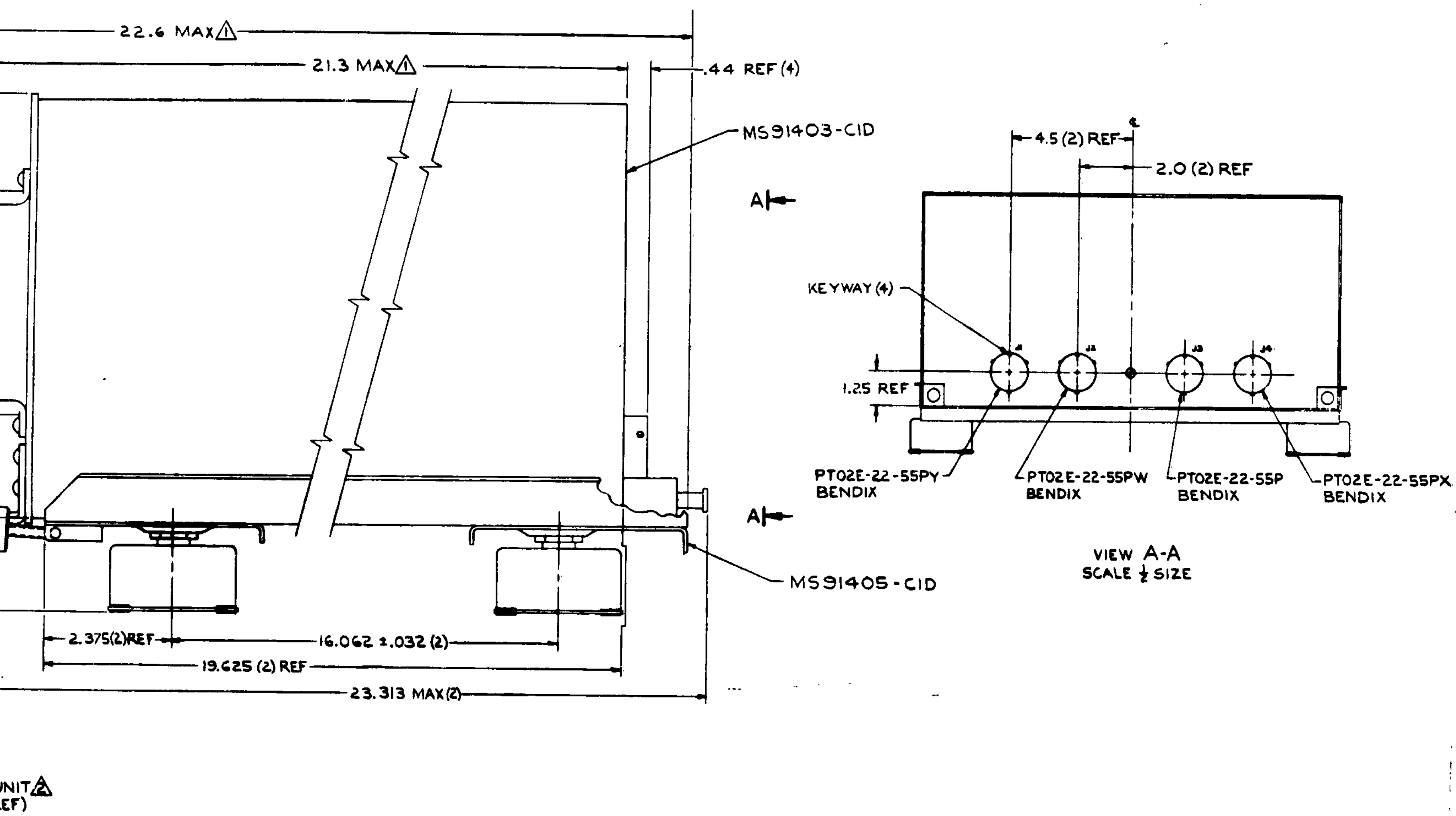


Figure 15. DUG 1331 S



UG1331 Switching Unit

2

Environmental Requirements

The environmental requirements for both the control unit and switching unit are given below.

High Temperature - This test shall be conducted in accordance with Procedure III (Par. 4.1.3) of MIL-E-4970A.

Low Temperature - This test shall be conducted in accordance with Procedure III (Par. 4.2.3) of MIL-E-4970A except that the 72-hour soak at -62°C shall not be performed.

Low Pressure - This test shall be conducted in accordance with Procedure II (Par. 4.4.1.2) of MIL-E-4970A except that the equipment shall be operated at 20,000 feet instead of 10,000 feet as specified.

Vibration - This test shall be conducted in accordance with Procedure VI (Par. 4.6.8) of MIL-E-4970A except that lower frequency range shall be 10 to 20 cps instead of 2 to 20 cps as specified, and the acceleration in the 52 to f_p range shall be ±3 g instead of ±5 g.

Radio Noise - This test shall be conducted per the applicable sections of MIL-I-6181D.

/G1352 BENCH MOCKUP

The DUG1352 Bench Mockup is contained in two steel 19-inch racks and is designed to provide both a sloping-front operator's panel and a horizontal surface for mounting the MH-96 adaptive calibrator. Provisions are made for monitoring all test connector voltages and selected system connector voltages. The bench mockup connects to the MH-96 system through five system cables and four test connector cables. The mockup is caster-mounted for mobility in the laboratory.

Appropriate loads are provided for outputs of the adaptive calibrator, and input power to the calibrator is provided through the bench mockup. All system sensors may be simulated by the mockup either singly or in pairs to supply the redundant inputs. Actual sensors may be connected when they are available for stimulus generators.

A minimum of metering is provided because this tester is not used after the completion of the environmental and life tests. Because the bench mockup will only be used in controlled laboratory-type environments, no stress testing will be performed and the mockup will be constructed of commercially available parts rather than government-source inspected components.

The DUG1352 Bench Mockup will provide the following in the manual mode:

1. Access to all connector pins and test points.
2. Adjustable steady-state a-c and d-c signals to all channels of one axis at a time.
3. One a-c and one d-c source for signals to be used on axes other than the one under test in (2).
4. Visual indication of interlock circuitry operation.
5. Loop closure around servo amplifiers simultaneously.
6. Aircraft simulation for three channels simultaneously.
7. The following types of gain measurements:
    a. Open loop
    b. Closed loop
    c. Fixed gain
    d. Minimum gain
8. Visual indication of disengage circuitry operation.
9. Differential servo-amplifier current measurement.

DUG1345 ACCELEROMETER TESTING BOOM

The DUG1345A1 Accelerometer Test Boom Fixture is a depot or field level fixture originally designed to check the linearity, hysteresis, and cross-coupling of the DGG182A1 Normal Accelerometer and DGG183A1 Lateral Accelerometer. It is a black anodized aluminum alloy fixture approximately 40.02 inches long, 7.62 inches high, and 7.01 inches wide.

A removable bracket at one end of the boom is used for mounting and positioning the accelerometers for their various testing modes. The bracket is so doweled that its position for both positive and negative g's has its center of gyration at 17.500 inches. On the opposite end of the boom a fixed counterweight is located for dynamic balance of the boom.

Four retractable thumbscrews are used to mount the boom on a Genisco or equivalent rate-of-turn table. The boom is accurately centered by a retractable centering plug which fits into a counterbore on the rate-of-turn table.

Electrical connections to the accelerometers are made through individual cables, depending on which accelerometer is being tested. One end of the cable is connected to the mounted accelerometer and the other is connected to the connector on the boom. A 17-inch seven-lead cable is connected to binding posts of the rate-of-turn table. These binding posts, in turn, go through slip-rings to other binding posts on the console for proper electrical connection to the accelerometers.